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Report No. NR81H-50

Rockwell International

SUMMARY REPORT

XFV-12A DIAGNOSTIC

AND DEVELOPMENT

PROGRAM

CONTRACT NO. N00019-73-C-0053

NOVEMBER 1981

North American Aircraft Division
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FORM 354-K REV. 3-77

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ABSTRACT

This report summarizes the diagnostic and development program of
the XFV-12A thrust augmented wing technology prototype conducted
during the period of January 1979 through May 1981 and is
presented in compliance with Item 0014 of Contract No.
N00019-73-C-0053.

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TABLE OF CONTENTS

Rockwell International

TABLE OF CONTENTS (CONTINUED)

![Rockwell International logo] Rockwell International

TABLE OF CONTENTS (CONTINUED)

Rockwell International

LIST OF FIGURES

LIST OF FIGURES (CONTINUED)

LIST OF FIGURES (CONTINUED)

LIST OF FIGURES (CONTINUED)

Rockwell International

LIST OF FIGURES (CONTINUED)

LIST OF FIGURES (CONTINUED)

Rockwell International

LIST OF FIGURES (CONTINUED)

Rockwell International

LIST OF TABLES

Rockwell International

LIST OF ABBREVIATIONS AND SYMBOLS

A	Area
A_2/A_o	Ejector Inlet Area Ratio
A_3/A_2	Ejector Diffuser Area Ratio
AR	Aspect Ratio
Asym	Asymmetric
B	Base Angle
BTR	Bow Tie Ratio
C	Dissipation Constant
C_1, C_2, and C	Turbulence Constants
C_D	Flow Coefficient $C_D = \dfrac{\text{Measured Weight Flow}}{\text{Ideal Weight Flow}}$
C_E	Nozzle Pressure Expansion Factor
C_P	*Pressure Coefficient*
C_T	Thrust Coefficient $C_T = C_D \times C_V$
C_V	Velocity Coefficient $C_V = \dfrac{\text{Measured Thrust}}{\text{Isentropic Thrust}}$
C_k	Energy Constant
C_{ref}	Reference Chord Length
C-D	Converging-Diverging
CIVV	Compressor Inlet Variable Vane
CS	Cross Slot
CTA	Coanda Turning Angle
CTE	Coanda Turning Efficiency $(P_{t_{peak}}/P_{t_{nozzle\ exit}})$
DAR	Diffuser Area Ratio
DLI	Deck Launched Interceptor

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LIST OF ABBREVIATIONS AND SYMBOLS (Continued)

FTIT	Fan Turbine Inlet Temperature
G	Rate of Generation of Turbulence Kinetic Energy
H	Enthalphy
I	Entrainment Index
K	Log Spiral Constant
L	Launch Angle
L	Lift
L/H	Left Hand
L/W or L/D	Ejector Length to Throat Width Ratio
l	Nozzle Width
l/t	Nozzle Aspect Ratio
M	Mach Number
MFR	Mass Flow Ratio
Mod	Modified
N_1	Engine Low Rotor Speed
N_2	Engine High Rotor Speed
NPR	Nozzle Pressure Ratio ($P_{t_{nozzle\ exit}}/P_{amb}$)
P	Pressure
P_r	Prandtl Number
P_S	Static Pressure
P_T	Total Pressure
P&W	Pratt and Whitney Aircraft
R	Coanda Local Radius of Curvature
R	Gas Constant
R_T	Empirical Constant

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LIST OF ABBREVIATIONS AND SYMBOLS (Continued)

R_1	Internal Coanda Radius
R_0	Initial Coanda Radius
RCVV	Rear Compressor Variable Vane
RPM	Revolutions Per Minute
S	Coanda Running Length
S	Spacing
S_{throat}	Coanda Running Length to Augmenter Throat
T	Temperature
TAW	Thrust Augmented Wing
U_j	Jet Velocity Initial
U, V, and W	Time Averaged Velocity Components
Unmod	Unmodified
V	Velocity
V_P	Local Peak Jet Velocity
V_o	Nozzle Exit Velocity
V/STOL	Vertical/Short Takeoff and Landing
VTOGW	Vertical Takeoff Gross Weight
VTOL	Vertical Takeoff and Landing
W	Local Jet Width
W	Wedge Angle
X	Internal Nozzle Envelope
Y	External Coanda Shape Envelope
a	Speed of Sound
e	Base of Napierian Logarithms (2.71828)

k	Turbulence Kinetic Energy
$\dot{m}$	Mass Flow
t	Jet Width Initial
t	Nozzle Gap
ρ	Fluid Density
γ	Ratio or Specific Heats
σ	Source Strength
ε	Turbulence Rate of Dissipation
γ	Vortex Strength
ϕ	Augmentation Ratio (Lift/Isentropic Thrust)
δ	Hypermixing Angle
σ_ε	Turbulent Prandtl Number for ε
σ_k	Turbulent Prandtl Number for k
β	Velocity Profile Shape Factor
μ_t	Turbulent Viscosity

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LIST OF ABBREVIATIONS AND SYMBOLS (Continued)

<u>SUBCRIPTS</u>

avg	Average
bt	Boat Tail
C	Cross Slot
Cn	Coanda Nozzle
e	Entrainment
H	Hypermixing
isent	isentropic
L	Lift
P	Primary
Pn	Primary Nozzle
S	Span Slot
Sec	Secondary
0	At Primary Nozzle
1	At Ejector Inlet
2	At Ejector Throat
3	At Ejector Exit

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1.0 INTRODUCTION

The purpose of this report is to document the activity during the XFV-12A thrust augmenter development program conducted from May 1979 to June 1981. This phase of the XFV-12A program was initiated subsequent to the tethered hover testing of the XFV-12A conducted at NASA Langley Research Center and immediately following a series of wing augmenter diagnostic investigations. During the diagnostic investigations, it was confirmed that the established goal performance levels of $\emptyset = 1.50$ for the wing augmenter and $\emptyset = 1.30$ for the canard augmenter system were not achieved during the tethered hover test program.

The diagnostic tests also indicated that a major source of performance deficiency could be attributed to feed duct/augmenter component interface problems rather than the basic augmenter configuration. The interface problems seriously affected both the isolated performance and flow quality of the primary augmenter nozzles resulting in significantly lower levels of augmentation than originally expected.

After an extensive review of both the tethered hover tests and the wing diagnostic investigations by both Navy and Rockwell International personnel, a remedial program which is the subject of this report was initiated. The primary objective of this program was to systematically improve the performance of the XFV-12A aircraft's thrust augmenter lift systems. Major emphasis was placed not only on correcting the flow deficiencies identified during the wing augmenter diagnostic investigations, but to identify and incorporate any improvements in the augmenter design which would increase its performance potential. The basic rationale for this redevelopment program was established by a series of specific guidelines which are listed below:

- The thrust augmenter development was to proceed in a serial fashion beginning with the development and demonstration of the wing augmenter performance, to be followed by a similar program for the canard augmenter system. Finally, assuming adequate performance levels were demonstrated, the XFV-12A aircraft would be reassembled with the improved augmenter configurations for VTOL hover evaluation.

- An existing analytical method utilized for thrust augmenter performance studies was to be improved and correlated with experimental data.

- Model scale and full scale isolated component testing would be employed to correct the feed duct/augmenter component flow deficiencies identified during the wing diagnostic investigations.

- Scale models representative of aircraft augmenter designs would be utilized to develop and demonstrate desired performance goals.

- Continuous aircraft/augmenter design studies would be conducted to assure that candidate augmenter configurations were compatible with aircraft packaging constraints.

Since its inception in May of 1979, this development effort has been
progressing in a promising manner. To date, a number of the program's
objectives have been successfully completed. The completed objectives
include:

- Redevelopment and demonstration of adequate performance and flow
  quality on full scale aircraft augmenter components as discussed
  in Sections 6.0 and 9.0 of this report.

- Utilizing both analytical and experimental methods basic
  augmenter and nozzle design improvements were developed which
  increased the performance of a 0.20 scale aircraft representative
  wing augmenter from $\emptyset = 1.46$ to $\emptyset = 1.64$. This effort is fully
  discussed in Sections 7.0 and 8.0 of this report.

- Design studies were successful in not only integrating candidate
  augmenter configurations in the basic XFV-12A aircraft, but were
  integral in the evolution of the basic aircraft design as
  described in Section 4.0.

Unfortunately, the redevelopment effort was terminated in June 1981 prior
to successful completion of all of the stated program objectives.

The subsequent sections of this report will present a detailed discussion
of the XFV-12A augmenter remedial program and its current status. A more
detailed description of the XFV-12A aircraft and a discussion of the
significant events leading to the development effort will be presented.
This will be followed by a discussion of the augmenter design studies and
the evolution of the aircraft design. The ensuing sections will then
describe the actual remedial development effort, its philosophy, the
facilities and methods utilized, and a presentation of significant
results. Finally, in the last section, a series of recommendations for
further development of the thrust augmenter/V/STOL aircraft technology
will be discussed.

2.0 BACKGROUND

2.1 GENERAL

On November 15, 1971, the Naval Material Command issued a "letter type" request for proposal to develop and test a V/STOL (Vertical/Short Takeoff and Landing) Technology Demonstrator Aircraft. Proposal guidelines specified that the V/STOL configurations address either a subsonic multipurpose mission or a supersonic fighter attack mission for air superiority or offensive fleet actions. The proposed V/STOL demonstrator aircraft should also be compatible with operations on the new smaller Sea Control Ship carrier concept. Additionally, the technology demonstrator concepts would incorporate imaginative ideas and be directed toward technology development rather than being restricted by established full scale engineering development constraints. Rockwell International responded to this request by submitting two proposals for aircraft utilizing the Rockwell thrust augmented wing concept (see Figure 2-1).

In October 1972, the Navy selected Rockwell International as the winner of this competition and contracted with Rockwell to develop a supersonic V/STOL fighter/attack aircraft employing the thrust augmented wing concept. This aircraft was later officially designated the XFV-12A.

The thrust augmenter wing concept (TAW) possesses several unique characteristics which identify it as a viable V/STOL aircraft lift system highly compatible with naval shipboard operations.

The low temperature and pressure footprint of the TAW concept is an obvious advantage when compared to other candidate V/STOL lift systems as shown in Figure 2-2. By virtue of entraining and mixing secondary ambient airflow in quantities several times the aircraft's propulsion system primary flow, both the exhaust pressure and temperature at the augmenter exit plane are reduced to 320 psi and 275°F, respectively. At these conditions, support personnel can operate effectively and safely near the aircraft during V/STOL operations.

The propulsion match between the augmenter and conventional mission requirements also tends to reduce the penalties normally associated with V/STOL aircraft propulsion systems. The net effect of the thrust augmentation including duct losses and control allowances allows the use of a smaller thrust engine than would be required for a deflected thrust concept. This results in a better subsonic thrust-to-weight cruise efficiency for the TAW concept. Figure 2-3 compares TAW with other V/STOL propulsion system concepts in a study conducted at the David Taylor Naval Ship Research and Development Center (DTNSRDC) for an aircraft sized to 35,000 pounds Vertical Takeoff Gross Weight (VTOGW) Deck Launched Intercept (DLI) mission. The comparison of percent useful load versus thrust/weight uninstalled shows the potential propulsion system benefits associated with the TAW concept.

The TAW concept also provides a positive circulation lift effect during conversion from vertical to conventional flight modes. As shown in Figure 2-4, this circulation lift provides increased total aircraft lift during this critical flight transition mode. This advantage is inherent to the

Fighter/Attack

Multimission

NAVY
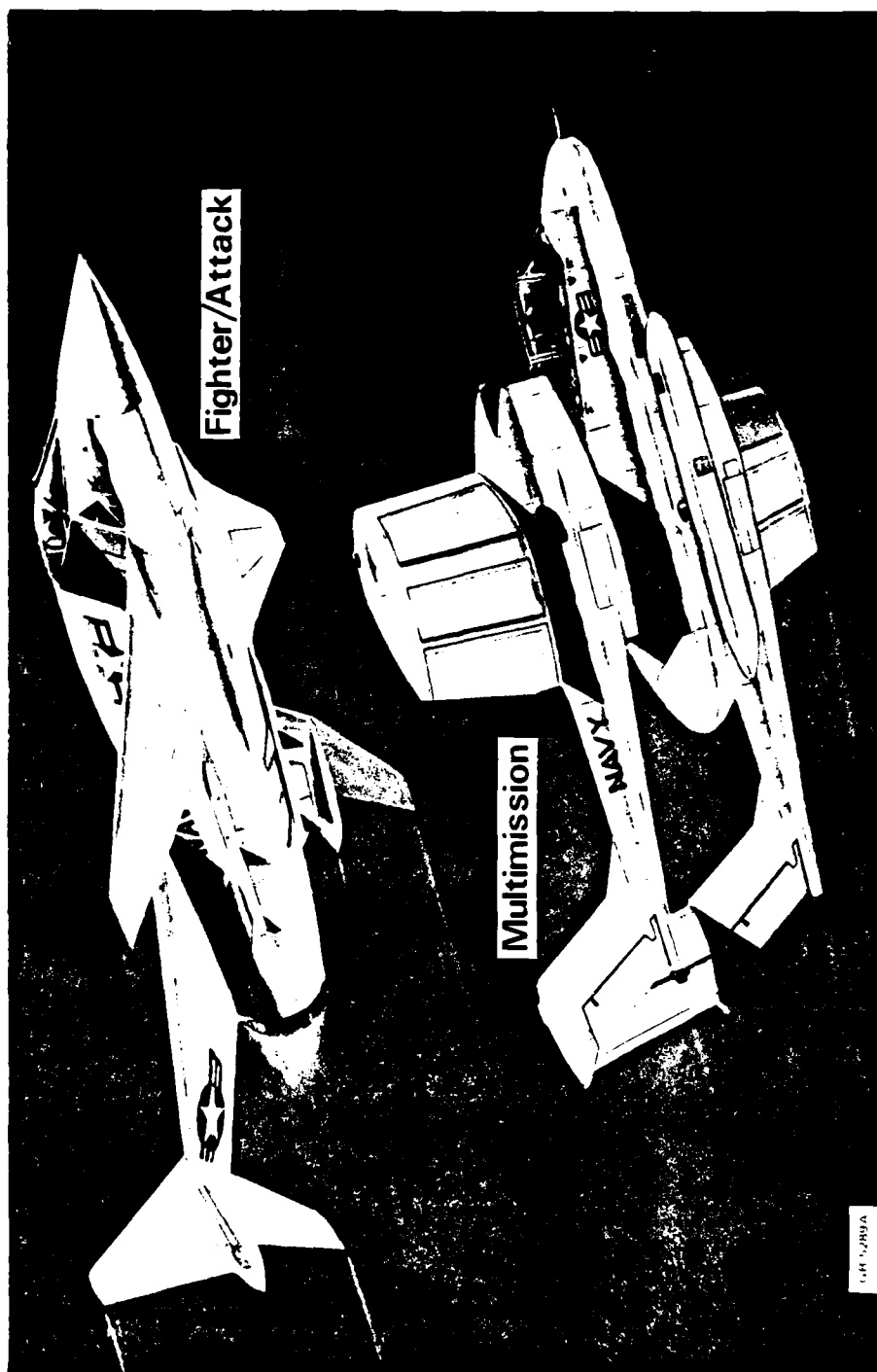
Figure 2-1 V/STOL Prototypes

LIFT SYSTEM FOOTPRINT
(AT EXIT)
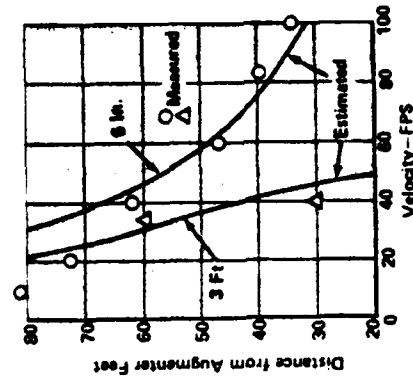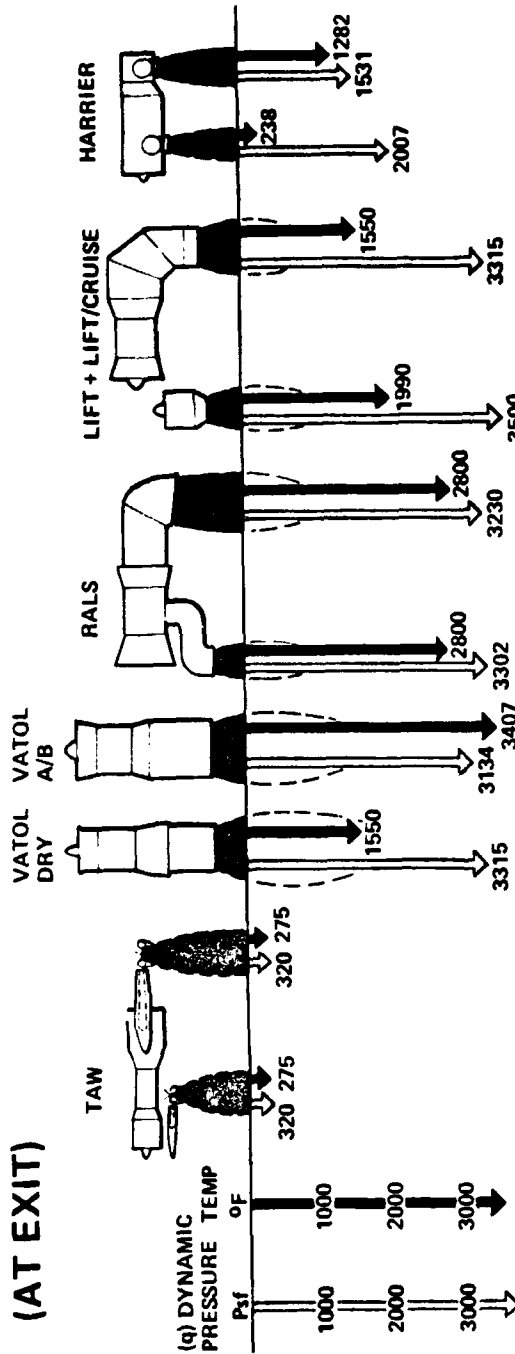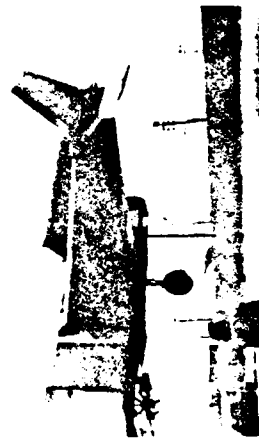
Figure 2-2   Lift System Footprint   (At Exit)

FIGHTER/ATTACK USEFUL LOAD COMPARISON

CAD 11195
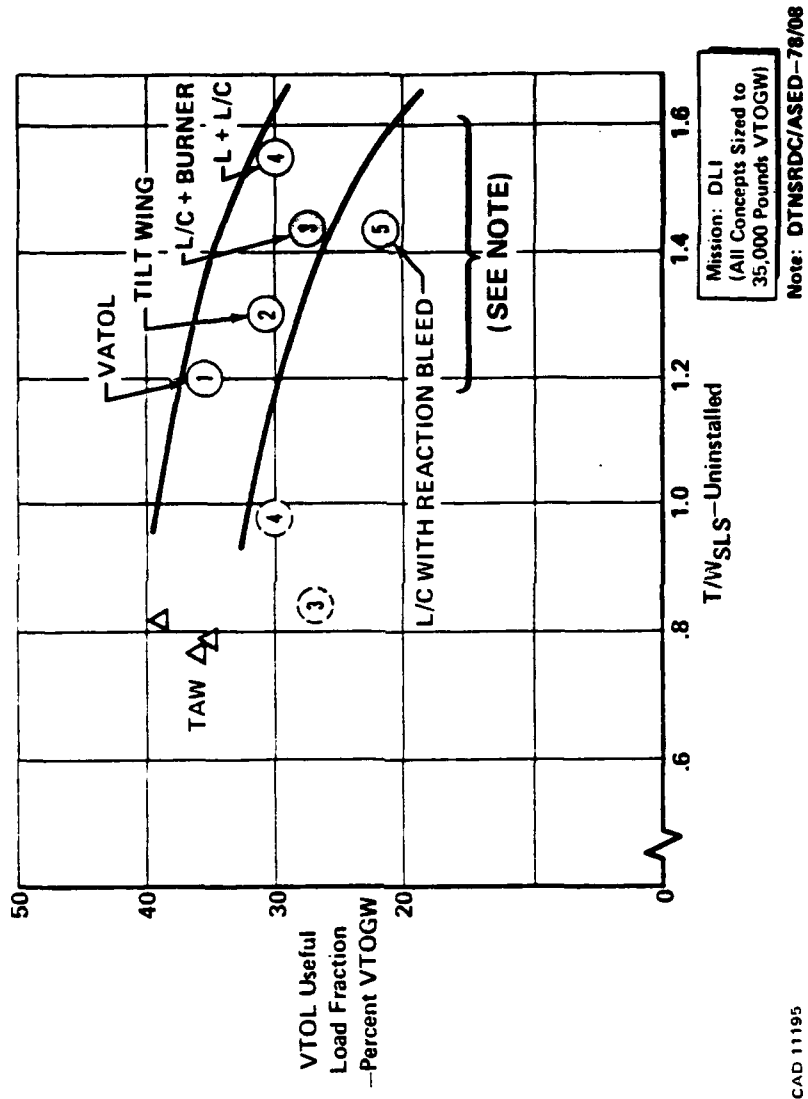
Figure 2-3 Fighter/Attack Useful Load Comparison

2-4

LIFT CONTROL Conversion
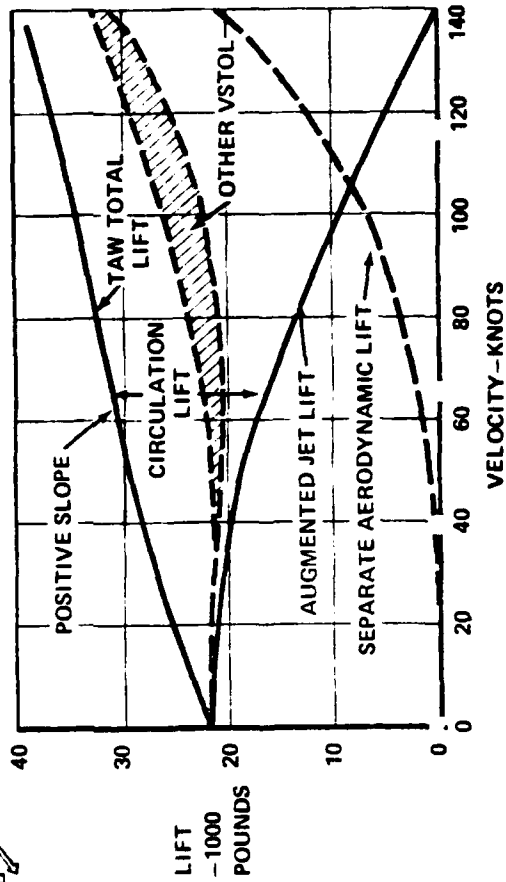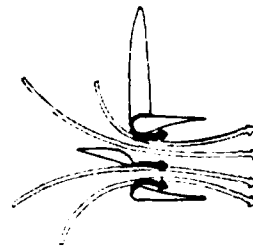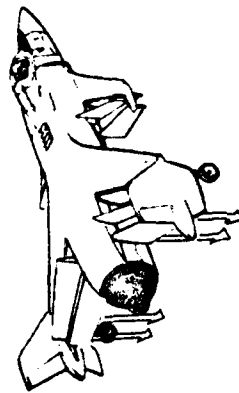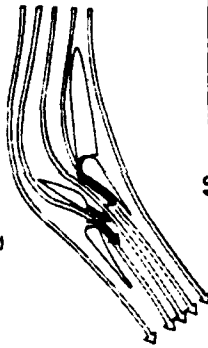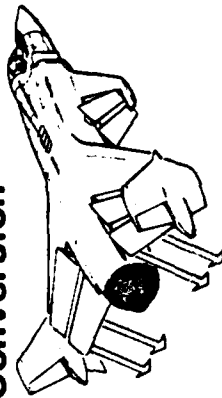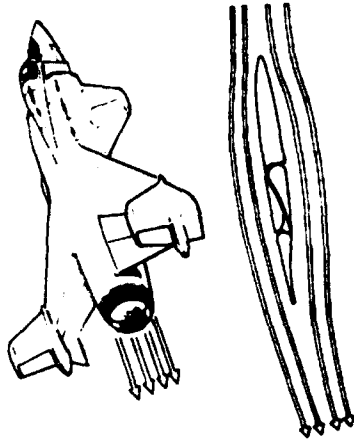
Figure 2-4 Lift Control Conversion

CAD 2282C

TAW concept because of its unique application of the aircraft (augmenter)
flaps as a portion of the aircraft's propulsion/control system.

2.2 AIRCRAFT DESCRIPTION

The XFV-12A Technology Prototype airplane is a single-place, single-engine
configuration that features the thrust augmented wing concept to achieve
vertical lift. The XFV-12A external configuration and arrangement are
shown in Figures 2-5 and 2-6. Aircraft dimensional data are presented in
Table 2-1. The configuration employs a low-canard, high-wing arrangement
with wing tip mounted vertical tails. The use of a canard reduces the
wing area required for maneuverability since both surfaces are always
lifting and makes feasible the use of wing tip mounted vertical tails.
The wing tip vertical tails serve a triple function of providing
significant increases in longitudinal stability and effective wing aspect
ratio as well as providing the basic directional stability and control for
the aircraft. The increases in longitudinal stability and effective wing
aspect ratio allows reductions to be made in wing span further compacting
the overall aircraft configuration. The multiple interactions of the
canard and vertical tails with each other as well as with the wing also
allow significant configuration aerodynamic refinement through minor
modifications to the basic aircraft arrangement. The primary engine
inlets are located on each side of the forward fuselage. An auxiliary air
inlet located in the upper forward fuselage aft of the cockpit provides
additional air during the V/STOL mode of operation (see Figure 2-7).
Every effort was made to reduce aircraft development costs through the
extensive use of assets from existing aircraft designs (see Figure 2-8).

Propulsion is provided by a Pratt & Whitney Aircraft YF401-PW-400 gas
turbine engine. In the XFV-12A aircraft, this engine is equipped with a
translating plug nozzle, shroud, and diverter door system (see
Figure 2-9). This diverter unit replaces the conventional aircraft
tailpipe and provides three basic modes of operation, an engine divert
mode (vertical), a forward thrust mode (conventional) and a transition
mode. The engine divert mode of operation directs the entire engine gas
flow output through a series of diverter openings into the aircraft's
ducting system where it is delivered to the augmenter system nozzles. The
engine's forward thrust mode of operation directs the engine gas flow
output aft through a translating shroud plug nozzle. In this mode, the
engine operates conventionally as a fixed tailpipe area engine. The
transition mode of operation schedules the engine gas flow through the
diverter openings and plug nozzle as required to maintain essentially
constant and uniform back pressure to the engine. This mode is utilized
to provide a smooth conversion from the vertical mode to the conventional
flight mode and return to the vertical model.

The aircraft augmenter ducting system is comprised of a series of thin
gage, complex shaped ducts formed from high temperature titanium alloys
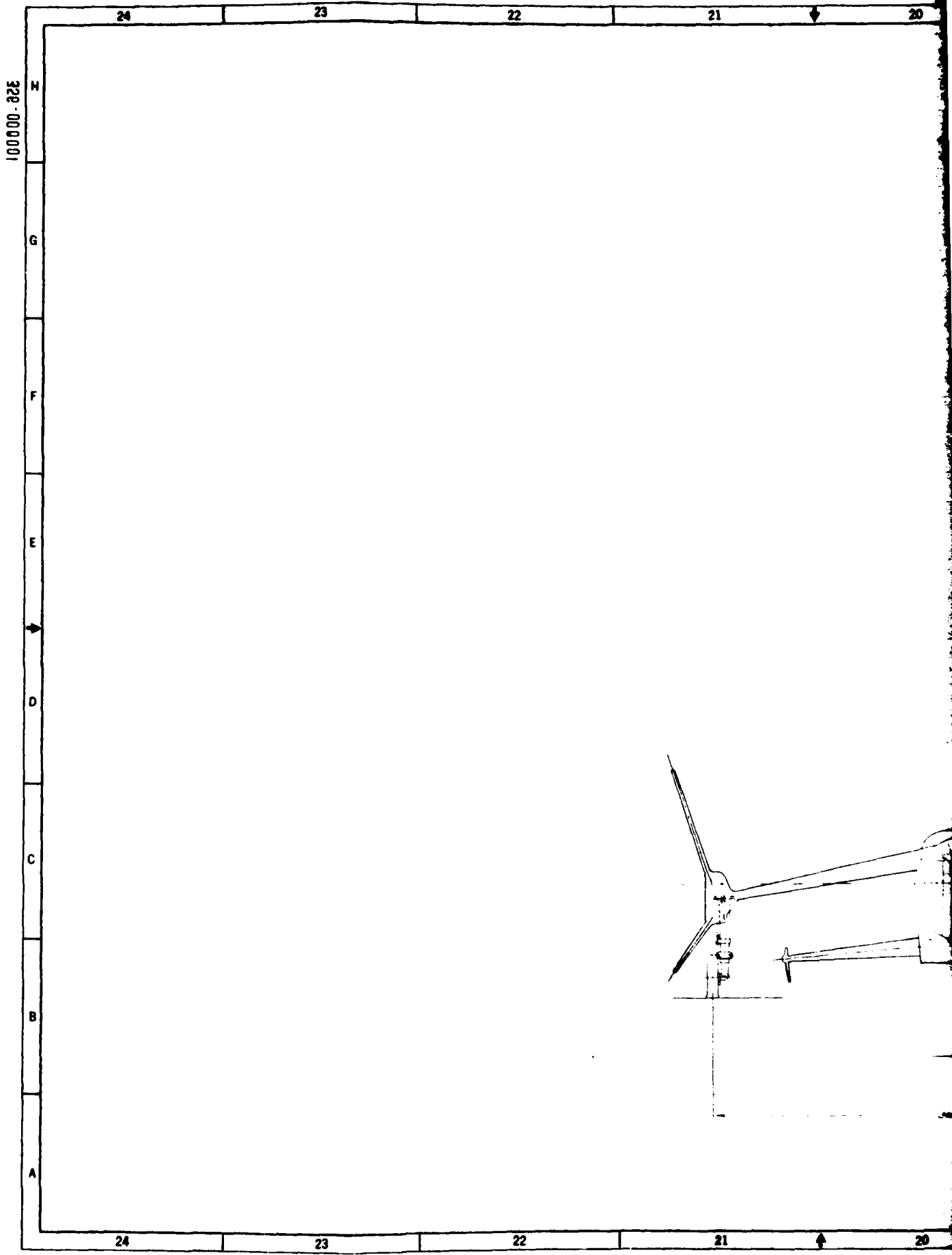
FUS AUXILIARY INLET DOOR

229.32
18.61 Ft.

CANARD DIFFUSER
HINGE LINE

CENT
HIN

F S
0

F S
108.0

F S
40.0

PILOT EYE

FS 111.62
WP 20.5

FUSELAGE REFERENCE PLANE
VERTICAL REF LINE

M 144.50

FRD
10° TRUE

5°

124.50 IN (10.37 FT)
OVERALL STATIC HEIGHT

tc 37.50
Er 95.00

COMP WL-78.07
STATIC WP 07.07
LATG WP 96.32

AR 151.250

205.50 IN (17.21 FT)

F S 122.41 (COMP)
F S 122.91 (STATIC)
F S 122.20 (EXTENDED)

| | 356-000001 | A | |

AFT TETHER ATTACHMENT

FS 343.15

FS 304.86

AIRCRAFT AFT HOIST POINTS

FS 270.8

₵ OF AIRCRAFT

344.9 IN. (28.74 FT)

BR 63.5

AIRCRAFT FWD HOIST POINTS
AND FWD TETHER ATTACHMENT

25 % CHORD
M.A.C.

WING FORWARD DIFFUSER FLAP
HINGE LINES (BASELINE AND T9)

WING CENTER EJECTOR
HINGE LINE

ELEVON AFT DIFFUSER
HINGE LINE

CENTER EJECTOR
HINGE LINE

ELEVATOR AFT DIFFUSER
HINGE LINE

FS
400.0

FS
300.0

616.3 IN. (51.358 FT)

120.8

FS 4.475

1.85°

F.R.P.

118.0 IN. (9.83 FT)

₵ ENGINE

10°

COMP. FS 62.594

STATIC FS 62.613

MAX TAIL DOWN GROUND LINE

6° 10'

TETHER TIE DOWN

FS 62.606 EXTD

STATIC GROUND LINE
STRUTS & TIRES STATICALLY DEFLECTED

GROUND LINE WITH MAIN GEAR STRUTS COMPRESSED WITH TIRES FLAT
AND NOSE STRUT COMPRESSED & NOSE WHEEL TIRE FLAT

303.93 IN. (25.49 FT) (STATIC)

NOTE
TABULATED AIRCRAFT DIMENSIONAL DATA CAN BE FOUND
IN REPORT NR-72H-330-1 SECTION 3.0

356-000001 | A

NR81H-50

XFV-12A AIRPLANE THREE VIEW

FIGURE 2-5

PARTS LIST

GENERAL ARRANGEMENT
THREE VIEW
NR 356

SIZE	CODE IDENT NO	
J	89372	356-000001

Z-7 / Z-8

5

DIFFUSER AIR MANIFOLD — CENTER EJECTOR AIR
AIRPLANE — ELEVON AIR

FRP
€ ENG

STRUCTURE

ENGINE DIVERTER OPENING
TIE BARS
CANARD PLENUM

SECTION G • G
FUS. STA. 438.9
AT € ENGINE DIVERTER OPENING
LOOKING FWD

AIRPLANE — FUEL
WING CENTER SECTION
(F-4 MODIFIED)

€ ENG — ENGINE ENVELOPE
FRP

PRESS REFUELING
RECEPTACLE

ENGINE REMOVAL DOOR

CANARD AIR DUCTS

SECTION F • F
FUS STA 35226
LOOKING FWD

MISSILES
(SPACE PROVISIONS ONLY)

WING TO FUS FWD ATTACH

AIRPLANE

HYDRAULIC PUMP

FRP
€ ENG.
HYDRAULIC PUMP

CANARD AIR DUCTS

STARTER

CONSTANT SPEED DRIVE ALTERNATOR

MISSILES
(SPACE PROVISIONS ONLY)

SECTION E • E
FUS CANT STA 302.0
LOOKING AFT

36.20

62.30

RUDDER

17.00

ⓒⓒ
S_UPPER 31.8
AR_UPPER 339
AIRFOIL

64.00

FUSELAGE AUX INLET DOORS — € AIRPLANE
DUCT AUX INLET LOUVRES

FUEL CELL

FRP
INLET DUCT

CANARD AIR MANIFOLD

CANARD

CANARD AUGMENTER
END PLATE DOOR

SECTION D • D
FUS STA 217.20
LOOKING FWD

AIRPLANE
MANUAL FLAP CONTROL — CANOPY
REAR VIEW MIRROR
PROPORTIONAL
DIFFUSER CONTROL — INSTRUMENT PANEL
CONTROL COLUMN

THROTTLE
LH CONSOLE
RUDDER PEDALS
R H CONSOLE
FRP

PILOT'S FLOOR

CABIN PRESS REG
DME EQUIP
DC CONVERTER

NOSE LANDING GEAR

SECTION B • B
FUS. STA. 111625
LOOKING FWD

ASN/75 COMPASS
€ AIRPLANE

SECTION A • A
FUS STA 6400
LOOKING FWD

WING SHOWN IN FUSELAGE REF SYSTEM

WING DATA

ORIGIN·FUS. STA 273.54,W.P.+220.61 & FUS. STA 400.0, W.P.+18.75

INCIDENCE	1°30
DIHEDRAL	-10°
S_{PROJ}	293.789 FT²
S_{ACT}	298.323 FT²
AR_{PROJ}	2.085
C_R	196.370 IN
C_T	88.516 IN

F.R.P

B.P 64.895 7 ζ=149.248

H REAR SPAR ON W.R.P

H FRONT SPAR ON W.R.P

₵ CENTER EJECTOR AXIS

₵ ELEVON AXIS

₵ DIFFUSER AXIS

LIFT CTR

38.20

M61A1 20 MM GUN
SECTION ₵₂ - ₵₂
FUS. STA. 191.0
LKG FWD

B.P 1455 ORIGIN LWR VERT
B.P 148.5 THEO WING TIP
B.P 1518 ORIGIN UPPR VERT

S_{UPPR} 31.9 FT²
AR_{UPPR} = .339
AIRFOIL NACA 64-004 (MOD)

75.00

19° CANT

₵ RUDDER HINGE .65 C

39°30

30° PLANE

35° PLANE

8400

INCIDENCE 3°30

84.00

6.13

ASN/75 COMPASS AMPL

₵ AIRPLANE

ASN/75 DIR GYRO

FRP
AIR SPEED UNIT (LOW SPEED V/STOL)
RADAR SET ADAPTER

MB-5A LOX CONVERTER (5 LITER)

SECTION A - A
FS STA 6400
OKING FWD

WING REF PLANE

B.P 1455

B.P 1518

S_{LWR} 927 FT²
AR_{LWR} = .771
AIRFOIL NACA 64-004

16.0

36.0

30° AB

30.2

35° CANT

THEO WING SECT
@ B.P 1518

POINT A
FS 419.65
W.P -22.83
B.P 153.75

OVERBOARD FUEL VENT
LH SIDE

FRP

WP-8203

| 356-000003 | H |

FIRE DETECTION CONTROL
FUEL FLOW POWER SUPPLY

RATIO CHANGERS
CONTROL SYSTEM MASTER ACTUATORS &
MIXER ASSEMBLY

ASN/75 DIR GYRO

EXTERNAL POWER
RECEPTACLE

AIR SPEED UNIT (LOW SPEED V/STOL)

C AIRPLANE

MB-5A LOX CONVERTER
RADAR ALTIMETER APN-94
ASN/75 COMPASS AMPL
BRAKE RESERVOIR
A4C FWD FUS SECTION
(TO FS 180.50)

F4 INLET
(MODIFIED)

VOR ANTENNA
ECS PRESS REG CONTROL
PRESS RATIO TRANSMITTER

C ELEVATO

C DIFFUSER AXIS   C CENTER EJEC
LIFT CTR

CANARD DATA

ORIGIN - C_aB WP-35.00, BP 37.50 & LE @ FS 133.00
INCIDENCE   = 0°
DIHEDRAL   = -5°
S_{EXP} = 82.7 FT^2_{ACT} · 82.385 FT^2_{PROJ}
AR   = 1.7654_ACT 1.7587_PROJ
B/2_ACT   = 72.50 IN
C_R   = 128.00 IN
C_T   = 36.26 IN
$\Lambda_{LE TRUE}$ = 35°46
AIRFOIL   = NACA 66-008375 (MOD)

FUS STA 994

SELF SEALING BLADDER DELETED

DOUGLAS ESCAPAC I-A1
ESCAPE SYSTEM
(ZERO-ZERO CAPABILITY)

PILOTS EYE POSITION
FUS STA 111.625, WP +20.50

HF ANTENNA

FUS STA 0   FUS STA 100   FUS STA 240 FUS STA 200   STA 2172   FUS

BRAKE SYSTEM RESERVOIR
AIR TRAFFIC CONTROL ANTENNA
AN-F1 EXT MANUAL CONTROL
(H.F. NEW)
ASN/75 COMPASS AMPL
AIRSPEED UNIT (LOW
SPEED V/STOL)
FRP
ASN/75 COMPASS
XMTR

INSTRUMENTATION
BOOM

RADAR ALTIMETER ANTENNAS
RADAR ALTIMETER APN-94
CABIN PRESS REG
DC CONVERTER

8203

30 REF

POINT B
FS 487.01
WP -23.27
BR 152.98

DME ANTENNA
CURRENT TRANSFORMERS (3)
THRU RH SIDE GENERATOR CONTROL UNIT
WHEEL WELL LINE CONTACTOR
AC/DC POWER DISTRIBUTION PANEL

EXTERNAL POWER RECEPTACLE

A4C NOSE LANDING GEAR

M-61A1 20 MM GUN
PACE DIVISION ONLY

RATIO CHANGERS
CONTROL SYSTEM MASTER ACTUATORS &
MIXER ASSEMBLY

STATIC DEC

PRESS FUELING RECEPTACLE

PNEUMATIC BOTTLE, MLG EMER REL

ELEVATOR AXIS

CENTER EJECTOR AXIS

FUSELAGE FUEL CELLS (420 GAL. TOTAL)

CANARD DUCT

BP 66888

FUSELAGE AUX. INLET SYSTEM

F-4 WING BOX (MODIFIED)

TOTAL WING FUEL (307 GAL)

HYDRAULIC RESERVOIR

FUSELAGE AUX. INLET DOORS

ANTI-COLLISION LIGHT
FUSELAGE FUEL CELLS

ENGINE FACE FS 292.8
FRONT ENGINE MOUNT FS 323.25

WING ACTUATING SYSTEM

ENGINE GROUND START
AIR CONNECTION
ENGINE ACCESSORIES
CANARD ACTUATING SYSTEM

ENGINE EXHAUST GAS DIVERTER

A4C MAIN LANDING GEAR

CANARD LOWER PLENUM

PRECOOLER

STATIC DECK LINE (6°10' TO FRP)

SPARROW III AIM-7F MISSILE (2) (SPACE PROVISIONS ONLY)
LOCATION: HORIZ W.P. -42.80 AT F.S. 323.25 & W.P. -11.35 AT F.S. 390.0
VERT. B.P. 20.40 AT F.S. 323.25 & B.P. 8.25 AT F.S. 390.0
MISSILE NOSE @ F.S. 245.6

356-000003

NR81H-50

XFV-12A INBOARD PROFILE

FIGURE 2-6

AIRPLANE

ZONE	LTR	DESCRIPTION	DATE	APPROVED
	G	REVISED AVIONICS IN NOSE COMPARTMENT & SECT A-A LINEAR ACTUATOR FOR WING WAS ROTARY ACTUATOR REVISED SECT G-G 31.8 FT² WAS 34.71 FT² (CANARD) REVISED AFT FUSELAGE LINES RELOCATED MISC EQUIP CHANGE NOT PREVIOUSLY RE-LEASED AT COLUMBUS	6-4-74	K JOHNSEN
	H	1 CHANGED AFT FUS UPPER MOLD LINES AND TAILCONE LINES. 2 CHANGED CANARD TIP LINES 3 CHANGED WING T.E.	DWG UPDATE	J. MARTINO 7-27-77

REVISIONS

ZONE	LTR	DESCRIPTION	DATE	APPROVED
		1 MAY BE RENEWED 2 CANNOT BE RENEWED 3 RECORD CHANGE 4 NON SHOP PRACTICE 5 PARTS MADE OR		
	A	REVISED	12-27-	
	B	REVISED & REDRAWN. ADDED SHT 2	5-3-73	
	C	REVISED FUSELAGE LINES AND ADDED NOTE 1, UPPER VERT WAS 27.506 FT, LWR VERT WAS 17.827 FT²	5-9-73	
	D	DELETED NOTE 1, UPPER VERT WAS 30.544 FT² MODIFIED AFT LINES. ADDED INTERNAL DETAIL INITIAL RELEASE, NC, A, B, &C NOT PREVIOUSLY RE-LEASED AT COLUMBUS	6-5-73	
	E	REVISED-OBSOLETED SHT.2 RELOCATED GUN MISSILE, HYD RESERVOIR, CONTROL MIXER, REVISED AUX INLET, FUEL CELL CONFIG, ENGINE DOOR (ONE PIECE) ADDED CANT. STA 302, REVISED OTHER FUS STRUC	9-14-73	WINTER
	F	INCREASED SPAN OF UPPER & LOWER VERTICAL TAILS. CHG'D ANGLE OF LOWER VERTICAL TO 35° ADDED FILLET TO AFT END OF M.L.G. POD.	3-13-74	JACKSON

CONVENTIONAL F401-PW-400 ENGINE
WITH A/B DOORS
OPEN
CLOSED

FRP
PWP-500
ENGINE

ENGINE EXHAUST
GAS DIVERTER

A4C MAIN LANDING GEAR

ER PLENUM

THIS ITEM REQUIRED FOR INSTRUMENTATION PURPOSES ONLY

FUS LINES AFT OF STA 3800 ARE NOT FIRM & ARE SUBJECT TO CHG

NOTES

ITEM OR FIND NO	QUANTITY REQUIRED	CODE IDENT	PART OR IDENTIFYING NUMBER	NOMENCLATURE	MATERIAL	THICK GAUGE MATERIAL	WIDTH OR I.D SIZE	L.G	DRAWING OR SPECIFICATION NUMBER	ZONE

PARTS LIST

			HEAT TREAT	UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES MFG TOLERANCES ON UNDIMENSIONED XX (DECIMALS) .03 XXX (DECIMALS) .010 ANGLES 30' BOLTS NOTED DRILL	CONTR NO	Columbus Division North American Rockwell COLUMBUS OHIO 43216
					DRN BY R. WINTER 12-8-72	INBOARD PROFILE GENERAL ARRANGEMENT XFV-12A
			FINISH		CHK BY	
					APVD	

ITEM	QTY	NEXT ASSY	USED ON	END ITEM NO	THRU	INSPECT PER MM 10870 (STO901LTO007 CLASS)	SIZE J	CODE IDENT NO 89372	356-000003
REQD PER END ITEM		APPLICATION		EFFECTIVITY		DO NOT SCALE PRINT	SCALE 1/10		SHEET

SCALE 1/10

CENTIMETERS

2-9 / 2-10

Table 2-1 Aircraft Dimensional Data

WING:

Symbol	Description	Value
S	Area (Reference)	293.0 Ft.²
S_{exp}	Area, Exposed	266.1 Ft.²
b	Span, Equivalent	24.5 Ft.
b_{th}	Span, Theoretical	25.68 Ft.
b_{exp}	Span, Exposed	19.0 Ft.
AR	Aspect Ratio	2.00
AR_{exp}	Aspect Ratio, Exposed	1.752
λ	Taper Ratio	.200
λ_{exp}	Taper Ratio, Exposed	.5151
Γ	Dihedral Angle	-10°
c_r	Root Chord	290.3 in.
$c_{r,exp}$	Root Chord, Exposed	11.5 in.
c_t	Equivalent Tip Chord	58.2 in.
$\bar{c}$	Mean Aerodynamic Chord	150.21 in.
Λ_{LE}	Sweep Angle (Leading Edge)	41.5°
$\Lambda_{.25c}$	Sweep Angle (.25 chord)	45.0°
i	Incidence Angle	+1.5°
$l_t/\bar{c}_w$	Rate of Taper	.714
S_v/S_w	Wing Length	.2969
θ_w	Aerodynamic Twist	0°

CANARD (Both Sides):

Symbol	Description	Value
S_c	Area, Exposed, Projected	7.65 Ft.²
b_c	Span, Exposed, Projected	12.05 Ft.
AR_c	Aspect Ratio, Exposed	1.764
λ_c	Taper Ratio, Exposed	.255
Γ_c	Dihedral Angle	-5°
$c_{r,c}$	Root Chord	125.4 in.
$c_{t,c}$	Equivalent Tip Chord	46.6 in.
$\bar{c}_c$	Mean Aerodynamic Chord	91.25 in.
$\Lambda_{LE,c}$	Sweep Angle (Leading Edge)	49.6°
$\Lambda_{.25c}$	Sweep Angle (.25 chord)	25.3°
i_c	Incidence Angle	0°
$l_t/\bar{c}_w$	Rate of Taper	1.466
S_c/S_w	Canard Area/Wing Area	.256

ELEVON:

Symbol	Description	Value
S_f	Area, Actual	20.52 Ft.²
$S_{f,proj}$	Area, Projected	20.25 Ft.²
b_f	Span, Actual	5.892 Ft.
$b_{f,proj}$	Span, Projected	7.265 Ft.
$c_{r,f}$	Root Chord	62.2 in.
$c_{t,f}$	Tip Chord	43.6 in.
$\bar{c}_f$	Mean Aerodynamic Chord	50.09 in.
c_f/c_w	Chord of Elevon/Chord of Wing	.15
$S_f \cdot \bar{c}_f$	Area Moment, Actual	51.5 Ft.³
λ_f	Rate of Taper, Actual	.705
$\lambda_{f,proj}$	Rate of Taper, Projected	.1692

ELEVATOR (One Side):

Symbol	Description	Value
S_e	Area, Actual	6.851 Ft.²
$S_{e,proj}$	Area, Projected	6.85 Ft.²
b_e	Span, Actual	6.100 Ft.
$b_{e,proj}$	Span, Projected	6.004 Ft.
$c_{r,e}$	Root Chord	45.64 in.
$c_{t,e}$	Equivalent Tip Chord	13.40 in.
$\bar{c}_e$	Mean Aerodynamic Chord	32.08 in.
A_e/A_c	Elevator Area/Canard Area (Projected)	.5566
c_e/c_c	Elevator Chord/Canard Chord-Averaged	.41
$S_e \cdot \bar{c}_e$	Area Moment, Actual	32.50 Ft.³
λ_e	Rate of Taper, Actual	.299
$\lambda_{e,proj}$	Rate of Taper, Projected	.441

Table 2-1 Aircraft Dimensional Data (Continued)

VERTICAL TAIL (One Side)

Description	Symbol	Value
Area, Projected (upper)	S_{vu}	27.12 ft²
Area, Projected (lower)	S_{vl}	15.52 ft²
Area (tot l)	S_v	43.14 ft²
Span, Projected (upper)	b_{vu}	5.125 ft
Span, Projected (lower)	b_{vl}	3.045 ft
Span, Projected (tot l)	b_v	5.173 ft
Aspect Ratio, Projected	AR_v	1.545
Taper Ratio (upper)	λ_{vu}	.526
Taper Ratio (lower)	λ_{vl}	.471
Root chord (upper and lower)	C_{rv}	54.0 in.
Equivalent Tip chord (upper)	C_{tvu}	44.2 in.
Equivalent Tip chord (lower)	C_{tvl}	40.4 in.
Mean Aerodynamic Chord (upper)	$\bar{c}_{vu}$	75.55 in.
Mean Aerodynamic Chord (lower)	$\bar{c}_{vl}$	75.12 in.
Leading Edge Sweep Angle (upper)	Λ_{vu}	49.5°
Leading Edge Sweep Angle (lower)	Λ_{vl}	52.75°
Area Moment (upper)	$S_{c_{vu}}$	17.27 ft²
Area Moment (lower)	$S_{c_{vl}}$	99.5 ft³
Vertical Area/Wing Area (2 verticals)	S_v/S_w	.294
Vertical length (upper)	l_{vu}/b_w	.447
Vertical length (lower)	l_{vl}/b_w	.496
Rate of Taper (upper)	α_{vu}	.64
Rate of Taper (lower)	α_{vl}	1.192
Cant Angle (upper)		19°
Cant Angle (lower)		10°
Incidence Angle (upper)	i_{vu}	4.5°
Incidence Angle (lower)	i_{vl}	0°

RUDDER (One Side)

Description	Value
Area, Actual	6.575 ft²
Area, Projected	6.217 ft²
Span, Actual	4.775 ft.
Span, Projected	4.569 ft.
Root chord	26.5 in.
Tip chord	16.0 in.
Mean Aerodynamic chord	21.25 in.
Rudder Area/Total Vertical Area (one side)	.191
Rudder Chord/Vertical chord	.85
Area Moment, Actual	11.66 ft³
Rate of Taper, Actual	.42
Rate of Taper, Projected	.576

HEATING

Description	Value
Length Overall	96.5 in.
Approximate Maximum Depth	? ft.
Baseline Surface Area	6.89 ft²

Figure 2-7 Engine Air Induction and Distribution System

Prototype Structure

A-4 Forward Fuselage

F-4 Wing Box

A-4 Main Gear

F-4 Inlet

A-4 Nose Gear
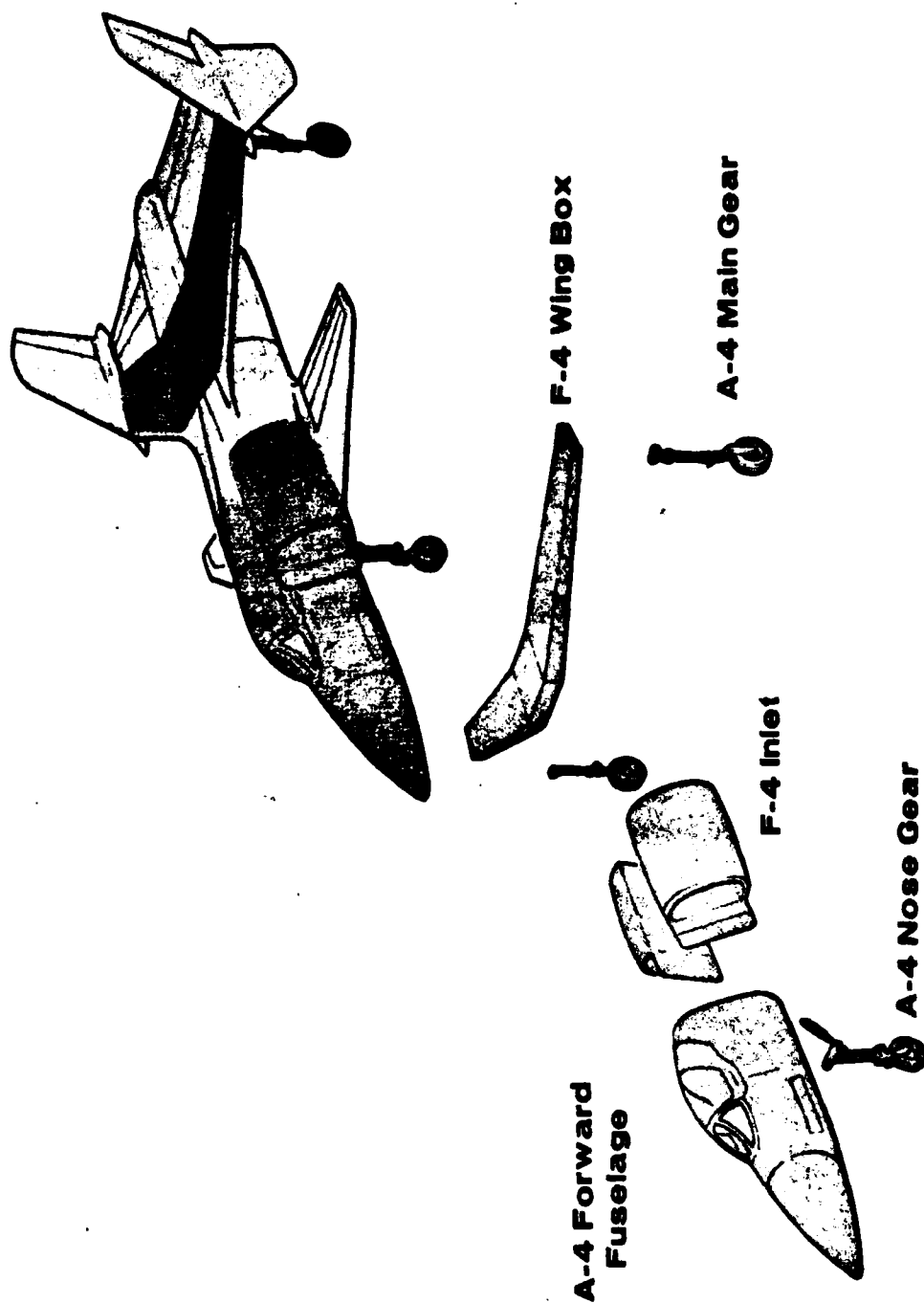
GR 4122D

Figure 2-8   Prototype Structure

DIVERTER VALVE DESIGN
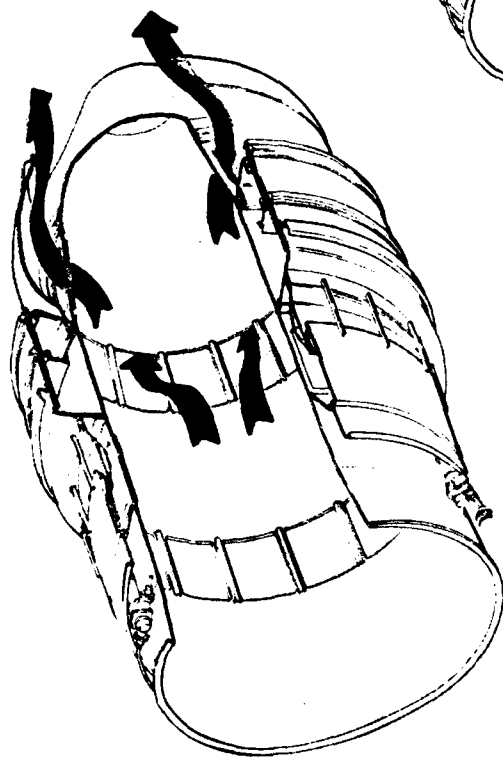
Vertical Mode

Horizontal Mode
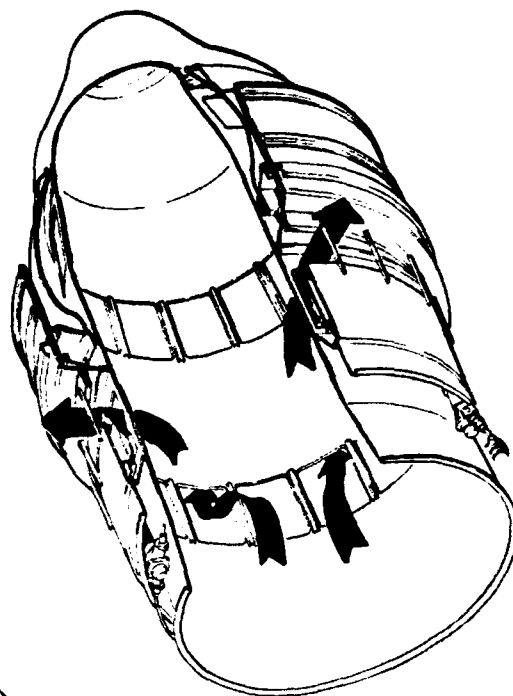
GR 1005-3

Figure 2-9 Diverter Valve Design

Rockwell International

with a minimum number of joints to reduce system weight. Conventional hot
forming techniques could not be employed to produce the complex shapes of
the ducting system without placing significant constraints on
configuration design options. Therefore, a superplastic forming process
utilizing a hot form chamber and low cost ceramic dies was developed.
This technique utilizes a unique temperature control process and radiation
shielding during the forming cycle permitting close tolerance fabrication
of complex shapes not possible with other fabrication methods.

The thrust required for the vertical operation is provided by the thrust
augmenting ejectors located in the aircraft's wing and canard surfaces.
Each ejector consists of a shroud formed by a pair of Coanda nozzle flap
components and a central plenum nozzle assembly as shown in Figure 2-10.
Entrainment by the primary exhaust jets is used to pump a large quantity
of air through the ejector shroud. By Newton's Law of Action and
Reaction, the ejector experiences a force which is equal but opposite to
the momentum change of the entrained air. A more complete description of
this process is given in Reference (1).

The lift generated by each augmenter can be simply modulated without
changing engine RPM by varying the exit area of the augmenter as depicted
in Figure 2-10. Height control is achieved by collectively controlling
the lift from all four augmenters. Pitch control is achieved by
differential lift generated by the canard and wing augmenters, and roll
control is achieved by differential lift generated by the wing
augmenters. Yaw control is achieved by differential control of left and
right wing augmenter mean angle.

Conversion to conventional flight as depicted in Figure 2-11 is
accomplished by gradual rotation of the augmenter flaps toward the closed
positions. The horizontal component of the rotated thrust vector provides
the force for the initial forward acceleration. During transition from
the vertical to the conventional flight mode, additional aircraft lift is
contributed by the very strong jet flap "supercirculation lift" effect
inherent to the TAW concept. During this transition mode, engine airflow
is rescheduled from the augmenters as a function of augmenter flap
position. When the augmenters are fully closed, the primary engine flow
is completely redirected through the cruise nozzle tailpipe.

In conventional flight, the augmenter flaps are retracted to form a
standard airfoil section and the aft augmenter flap on both the wing and
canard operates as an aerodynamic control device. Aft flaps on both the
wing and canard are used for pitch control and wing flaps are used
differentially for roll control. The fore and aft wing augmenter flaps
are used together as speed brakes, as depicted in Figure 2-10. Yaw
control in conventional flight is provided by left and right rudders.

LIFT AND CONTROL

AIRFOIL CONFIGURATION

CONVENTIONAL FLIGHT

SPEED BRAKE (Wing Only)

15°

25°

LIFT VECTORING

LIFT DECREASE

LIFT INCREASE

ROLL

Wing

Control Stick

YAW

Rudder

Wing

Rudder

Rudder Pedals

PITCH

Wing

Control Stick

Canard

V/STOL
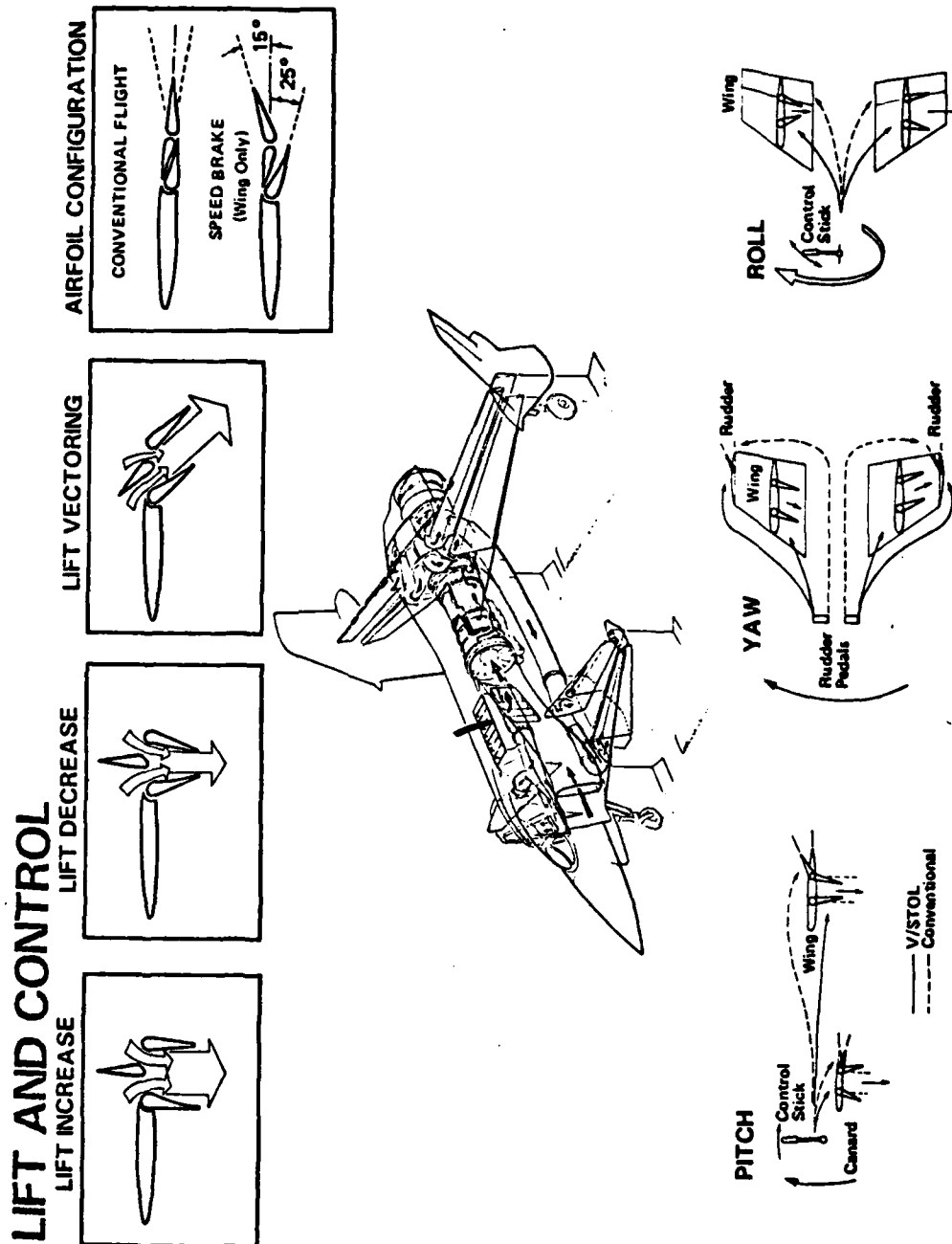
Conventional

Figure 2-10 Lift and Control

HOVER   STOL/CONVERSION   CRUISE
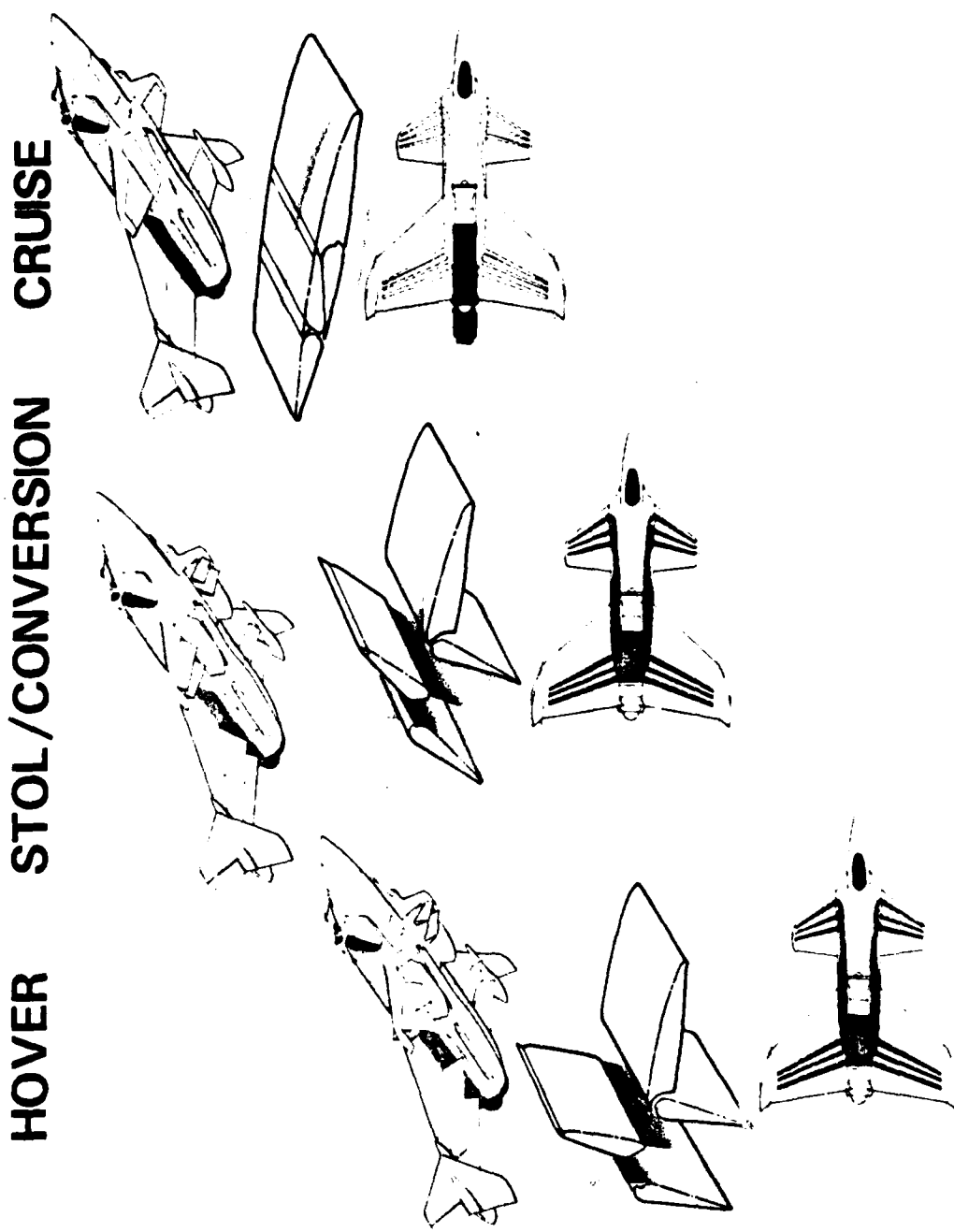
Figure 2-11   Conversion to Conventional Flight

GR 14330B

Rockwell International

2.3   XFV-12A AIRCRAFT DEVELOPMENT

The XFV-12A Technology Demonstrator Aircraft Program utilized both scale models and full scale hardware testing to develop the aircraft and associated systems configurations. Due to the unusual conceptual nature of this aircraft, five key areas of technical uncertainties were identified:

1. Flying qualities of the low canard, high wing configuration

2. Stability and control characteristics during the vertical and transition modes of operation

3. Compatibility of the exhaust diverter system with the P&W YF401 gas turbine engine

4. Efficient ducting of engine efflux to the thrust augmenter systems located in the wing and canard surfaces

5. Development of sufficient augmentation ratio by the thrust augmenter systems for effective VTOL operation

The issues of low canard/high wing flying qualities and stability and control characteristics were satisfied in the aircraft development phase and are fully reported in References (2) to (4). While the remaining technical issues were investigated during the development phase through the use of scale model testing, they could not completely be satisfied until the full scale aircraft could be tested, as discussed in the following sections.

2.3.1   Static and Dynamic Tethered Hover Tests

In 1977, the initial development and fabrication of the XFV-12A Technology Demonstrator Aircraft was completed. A series of functional and structural ground tests were performed at Rockwell's Columbus plant to validate the aircraft systems' operation and integrity. Following the ground testing, the aircraft was shipped to NASA Langley for the static and dynamic tethered hover test program. This program is documented in detail in Reference (5).

2.3.1.1 Facility

Th tethered hover testing utilized the Impact Dynamics Research Facility located at the NASA Langley Research Center in Hampton, Virginia. The Impact Dynamics Facility's 250 foot high "A" frame gantry provided structural support for the aircraft tether system (see Figure 2-12). The tether system was made up of a one-inch diameter cable, shock absorber, position sensor and a 50,000-pound capacity load cell. This assembly was suspended from a high line which was located on a platform on the gantry bridge. Figure 2-13 shows a schematic of the aircraft tether system. For static testing, the tether system was utilized in combination with a seven cable restraint load cell system, shown in Figure 2-14, to resolve
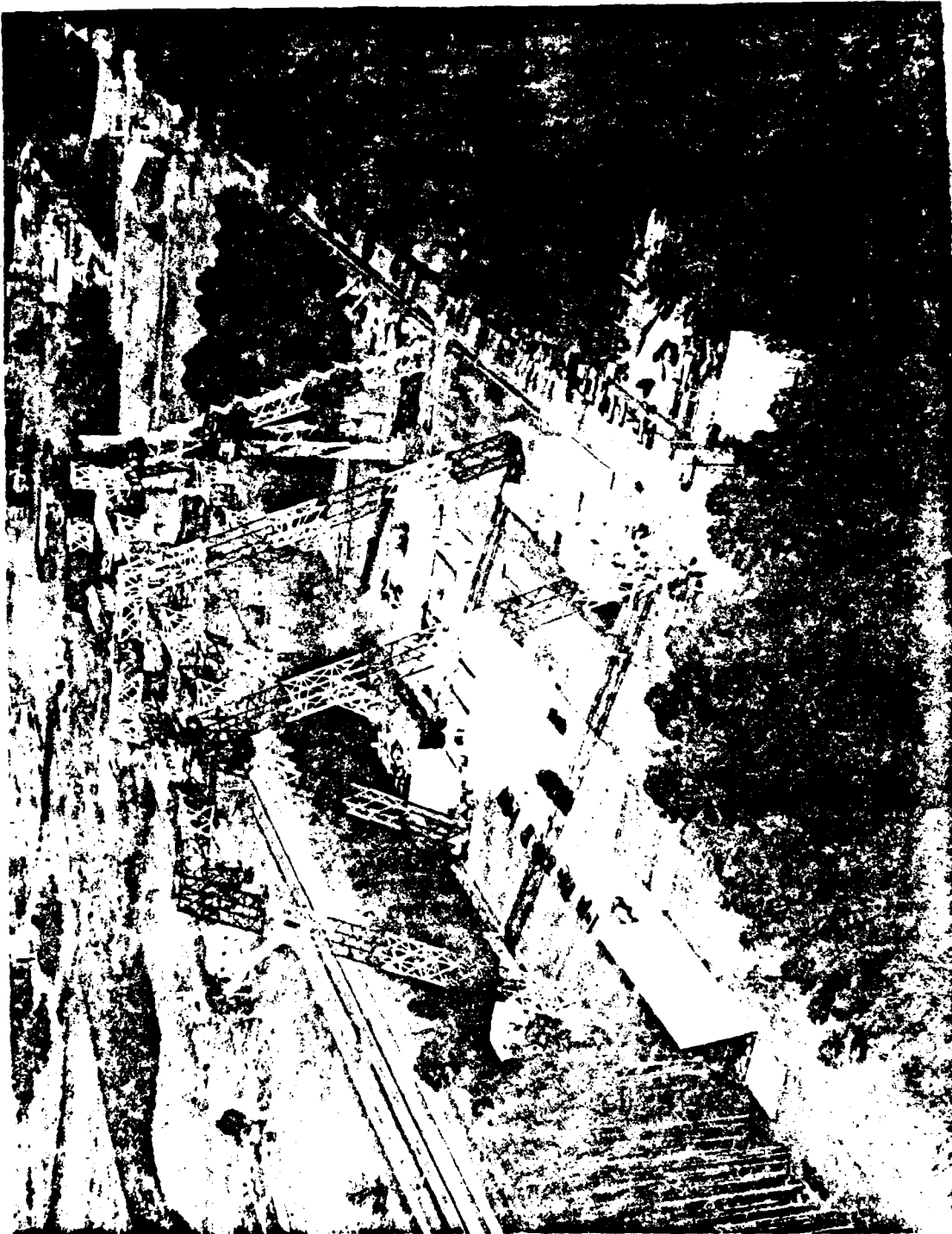
Figure 2-12 Test Site – Impact Dynamics Research Facility at Langley

Winch

1-inch Wire Rope

Open Spelter Socket
Axial Swivel
Universal Link

Shock Absorber

Universal Link

10-foot Cable Assembly
with Open Spelter Sockets

Universal Link
Position Sensor

Load Cell

Structural Interface

Umbilical

Shock Absorber

Load Cell

Structural Attach Unit

Position Sensor

Figure 2-13   Z-System

Load Cell Signals to
Aircraft Instrumentation

Disconnect

3 Components

1 Component
Load Cell

2 Components

2 Components

Figure 2-14 Static Load System

aircraft forces and moments. During dynamic testing, the aircraft was suspended on the gantry by the tether system. Prior to testing, the engine was started and the aircraft was positioned at a selected ground height by the high line winch assembly.

2.3.1.2 Test Program

The objectives of the tethered hover test program were to confirm the lift control and moment characteristics of the XFV-12A aircraft. The test program was broken into two distinct phases beginning with the static testing in order to isolate the force and moment characteristics followed by the dynamic test phase to demonstrate the aircraft's hover and control characteristics. Unfortunately, early in the static test program, it became apparent that the total measured lift of the aircraft was lower than was anticipated. Based on this information, the original test plan was modified to include several additional objectives.

-   Complete as much of the originally scheduled test as possible based on the limitations placed on the aircraft by insufficient lift for hover operations.

-   Utilizing available diagnostic instrumentation, identify and investigate causes for the reduced level of augmenter performance.

-   Based on the available diagnostic information, improve the performance of the thrust augmenter systems.

In order to accomplish these objectives, the test period was extended from the originally planned twelve weeks to a program of twenty-six weeks. During this period, a total of 83 static and dynamic test runs were completed. A complete listing of the test operations during the tethered hover test program can be found in Reference (5).

2.3.1.3 Results of the Tethered Hover Testing

The data obtained during the tethered hover testing provided a great deal of information concerning aircraft systems performance and integrity as well as satisfying a number of the original technical uncertainties. The following list summarizes the pertinent conclusions resulting from the tethered hover test program.

- Propulsion System

    The Pratt & Whitney YF401-PW-400 turbofan engine operated for a total of 28.77 hours (5.36 hours at intermediate power) during which no test time was lost due to engine related problems.

    No problems were identified with the exhaust diverter system either during engine trim or test operation.

    Inlet recovery data obtained during test operations indicated that the trifurcated inlet system achieved a 97.5 percent pressure recovery with relatively low pressure profile distortion.

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The exhaust diverter/flow ducting system pressure drops and
leakage rates compared well with pre-test estimates. Total
thrust loss was measured at 12.9 percent with 9.9 percent of this
due to total pressure loss and three percent associated with
system leakage losses.

In summary, the XFV-12A technology demonstrator propulsion system
from the trifurcated inlets to the ducting augmenter interface
locations was very efficient. Collectively, these findings
address and satisfy the technical uncertainties associated with
the compatibility of the YF401 turbofan with the exhaust diverter
system and the efficiency of the exhaust diverter flow ducting
system.

- ## Aircraft Hover Control

Test data obtained during the static test phase indicated that
control sensitivity and force gradients compared well with both
pre-test piloted simulations and the NAVAIR simultaneous power
requirements. Based on these correlations, the aircraft handling
qualities were considered satisfactory for dynamic hover testing.

Pilot workload during dynamic hover testing in and out of ground
effect was considered quite acceptable by the test pilots.

In general, the control system proved to be highly responsive,
thereby, partially addressing the technical uncertainties
concerned with the stability and control characteristics in the
hover mode.

- ## Aircraft Footprint

Augmenter exhaust temperatures were on the order of 250°F.

Augmenter exhaust velocities along the ground 50 ft. from the
aircraft were approximately 60 to 100 ft./sec.

These findings as well as the qualitative impressions of test
personnel around the aircraft during hover operations indicate
that the aircraft does possess a benign pressure-temperature
footprint, compatible with shipboard or confined area operations.

- ## Ground Effects

Variation of lift in ground effect correlated well with previous
scale model data indicating positive ground effect
characteristics at all heights measured.

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- Reingestion

 Several techniques were investigated which minimized the
 reingestion of thrust augmenter efflux. These techniques provide
 a means of controlling exhaust reingestion and indicate that no
 significant performance penalty will be realized due to efflux
 reingestion under operational conditions.

- Aircraft Structures

 During the entire aircraft tethered test program, only minor
 structural problems were noted. None of the problems compromised
 the basic structural integrity of the affected parts nor was
 there any damage to primary aircraft structure.

- Thrust Augmenter Systems

 Static lift (load cell) measurements obtained during the first
 phase of the tethered hover tests indicated that the performance
 of the thrust augmenter systems was significantly below the
 anticipated goal levels. As shown in Figure 2-15, the wing
 performance goal was $\emptyset = 1.50$, however, a $\emptyset = 1.26$ was achieved.
 The canard performance goal was $\emptyset = 1.30$ while a $\emptyset = 1.11$ was
 obtained.

 Based on these results, a thorough investigation of the thrust
 augmenter systems appeared to be the next logical step in an
 effort to identify the cause for the inadequate thrust augmenter
 performance.

2.3.2 Diagnostic Testing

The primary objective of the diagnostic test program was to identify the
causes of the inadequate levels of thrust augmenter performance during the
tethered hover test program. The decision was made to proceed first with
an investigation of the left hand wing augmenter system. The diagnostic
tests of this system included the following investigations:

 Testing of the left hand wing augmenter on the whirl rig test
 facility. These tests utilized aircraft augmenter and ducting
 hardware, in combination with simulated diverter, fuselage and wing
 leading edge components.

 Cold flow isolated component testing in the Thermodynamics Laboratory
 of

 - Forward diffuser (forward Coanda diffuser flap component)
 - Elevon (aft Coanda diffuser flap component)
 - Seven degree hypermixing centerbody segment model
 - Inboard and outboard endwall blowing nozzle assemblies

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CAD 10329
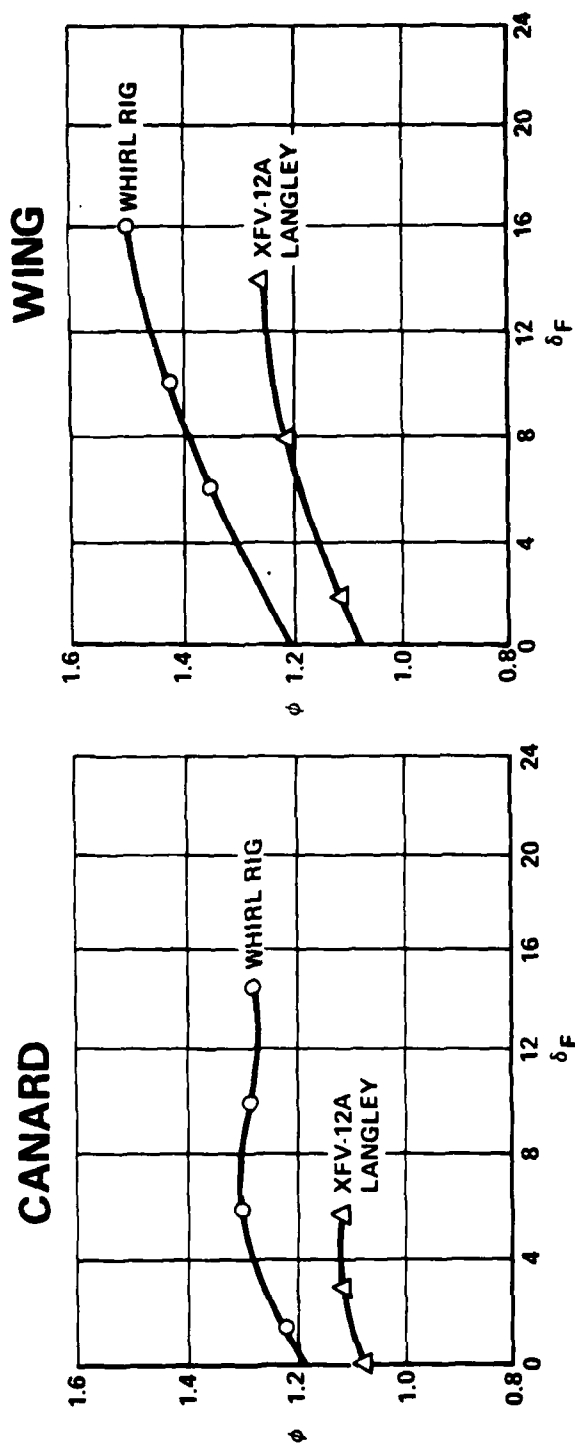
Figure 2-15 Augmenter Performance

Hot flow component testing on the whirl rig facility of the

- Forward diffuser flap (forward diffuser)
- Aft diffuser (elevon)
- Centerbody (Seven degree hypermixing nozzle and plenum assembly)

2.3 2.1 Diagnostic Test Results

A complete discussion of the diagnostics tests can be found in Reference (6)   Pertinent results are summarized below:

1. Test results of the full wing augmenter on the whirl rig correlated reasonably well with the performance levels obtained during the tethered hover testing indicating a maximum augmentation ratio of $\emptyset = 1.26$ for the wing augmenter system.

2. Isolated component testing identified numerous problems associated with each of the primary augmenter components.

   - Aft diffuser flap (elevon)

       - High degree of spanwise flow angularity
       - Irregular spanwise nozzle exit pressure distribution
       - Low component thrust coefficient

   - Forward diffuser flap (forward diffuser)

       - Irregular spanwise pressure distributions
       - Nozzle gapping under pressure and temperature

   - Centerbody Nozzle

       - Irregular spanwise nozzle exit pressure distributions
       - High degree of flow angularity/flow not directed uniformly over endwall

   - Endwall Blowing Nozzle

       - Irregular nozzle exit pressure distributions
       - High degree of flow angularity/flow misdirected.

2.4   Summary

The diagnostic tests confirmed that the goal performance levels of the wing augmenter system were not realized during the tethered hover testing. The primary cause of the inadequate performance was identified to be related to the flow quality and isolated performance of each of the primary components of the augmenter system rather than inherent to the basic augmenter configuration. These flow related problems did not exist on the wing development hardware and appear to be related to modifications incorporated in the aircraft augmenter components. These modifications

were necessary to integrate the augmenter system within the aircraft's
contours. The major difference resulting from these modifications was the
interface of the primary flow feed ducts with the individual augmenter
components. These differences seriously affected both the isolated
performance and flow quality of the primary augmenter nozzles. When these
inferior components were collectively operated as an augmenter, they
combined to significantly reduce the performance potential of the system.

After extensive analysis and review of both the tethered hover test and
diagnostic test data, a remedial thrust augmenter development program was
initiated. The primary objectives of this program were to develop in a
serial fashion the wing and canard augmenter systems performance allowing
a follow-on tethered aircraft hover test program.

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3.0 DEVELOPMENT PROGRAM

3.1 DEVELOPMENT LOGIC

Results of the diagnostic tests conducted on the XFV-12A Prototype No. 1 left-hand wing augmenter components identified the problems associated with each component of the wing augmenter system. For each of the defective components, an improvement rationale and method was identified. This program would rework all of the components and demonstrate on the whirl rig the initial desired performance, i.e., ϕ_{wing} = 1.50 and ϕ_{canard} = 1.30. This would allow more than ample test time for VTOL evaluation as shown by the lift summary in Figure 3-1.

A very detailed task oriented program was developed utilizing an analytical computational fluid dynamics program (developed using discretionary funds), reduced scale experimental testing, full scale component tests, and design studies of aircraft configuration constraints. Figure 3-2 shows the program as developed and the sequentially interlocked tasks. This program was organized in two phases. The first phase was oriented toward modifying the existing components to correct their individual discrepancies and demonstrate the original goal performance level. The objective of the second phase was to develop an advanced centerbody configuration and the necessary modifications to the augmenter geometry to demonstrate increased performance. Subsequently, the phase two performance level was accentuated in order to provide for a more comprehensive evaluation of the thrust augmented wing technology. Additional control capability (over that specified by MIL-F-8300) was desired during hover in order to assure adequate control response for aircraft evaluations in high gust level conditions representative of operations in the vicinity of small ships in high sea states.

This desire to obtain increased performance initiated an evaluation of hover characteristics and the performance requirements of the wing and canard configuration. This evaluation was based upon the configuration, as identified by ongoing design studies, which were required to incorporate an improved wing and a new canard augmenter. A new canard configuration had been identified with an estimated performance capability of ϕ = 1.40 for full scale hardware as installed on the XFV-12A. To obtain a wing ϕ that would support this higher canard ϕ, it was necessary to evaluate the geometric relationships of the wing and canard as well as the vertical takeoff gross weight. To provide for test flexibility in the augmenter nozzles and a substantial fabrication time and cost savings, corrosion resistant steel material was substituted for titanium material in the weight evaluation. An additional fuel reserve was carried in the weight estimate to provide for a heat sink in the fuselage fuel sump tank. This vertical takeoff gross weight (VTOGW) summarized to 19,003 pounds with five minutes of fuel at intermediate engine power.

SUMMARY OF EXPECTED TEST MISSION PERFORMANCE

LIFT SUMMARY

INSTALLED THRUST-16,608

TO CANARD	9,184
LOSS AND LEAK	-1,263
AT NOZZLE	7,921
X φ (1.306)	= 10,345
TO WING	7,424
LOSS AND LEAK	-839
AT NOZZLE	6,585
X φ (1.505)	= 9,910
MAX LIFT =	20,255 LBS
TRIM ALLOW	-625

MAX TRIMMED LIFT 19,630 LBS

*CONTROL AND
ACCEL
ALLOWANCE   -1,080

MAX VTOGW 18,550 LBS

*Meets Proposed NAVAIR Change to
Simultaneous Control Criteria in Hover.

TRIMMED BALANCED LIFT
(4° θ, 6° φ, 3° ψ)

VTOL GROSS WEIGHT

(Simultaneous
4° θ, 6° φ, 3° ψ)

Fuel Usage

385 LBS FUEL RESERVE

VTOL Test Time-Minutes

Lift/Weight
Pounds

CAD 11129C

WT EMPTY	15,685 (JUNE 1978 WEIGHING)
INST. AND TEST INSTALLS	1,111 (AS OF JUNE 1978)
	16,796
REMOVALS AND REDESIGN	-591
	16,205
PILOT, TRAPPED FUEL, OIL, ETC.	+484
	16,689 **OWE**

-395 ADDITIONAL REMOVALS IF REQUIRED

Figure 3-1   Summary of Expected Test Mission Performance
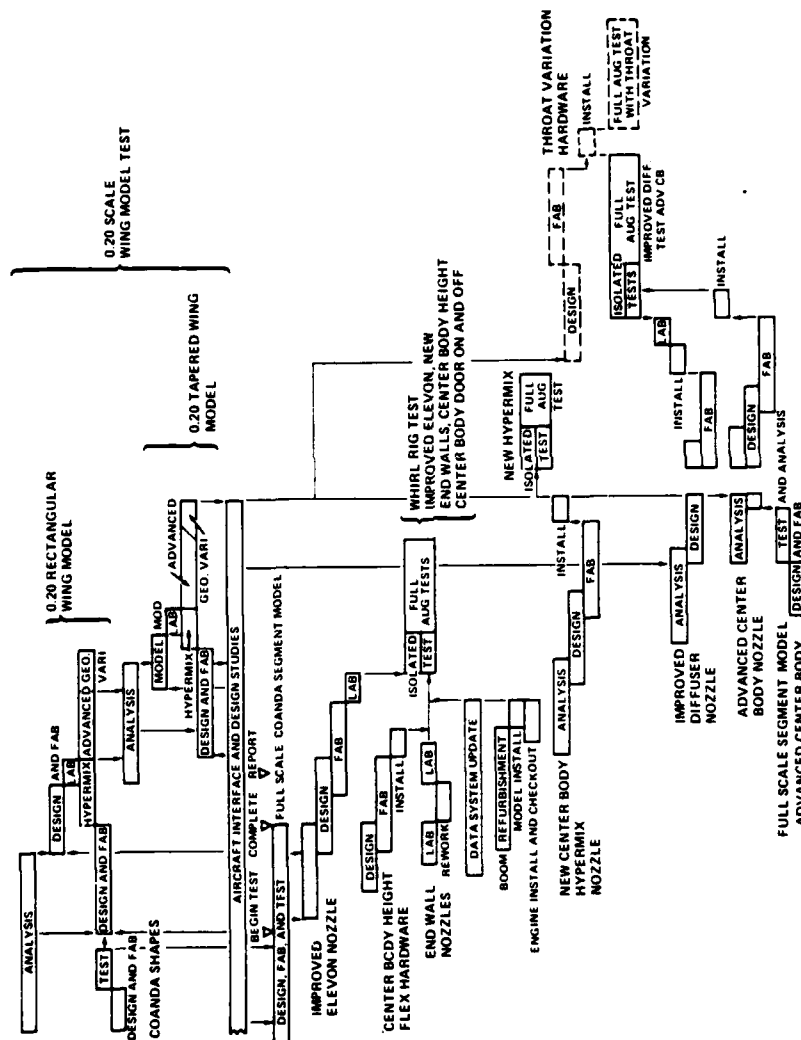
Figure 3-2   Wing Development Program

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For the geometric relationships of the wing and canard thrust vectors,
Figure 3-3, and the total nozzle thrust of 14,520 pounds (engine thrust of
16,608 less internal duct losses and leakage of 2088 pounds) a plot of
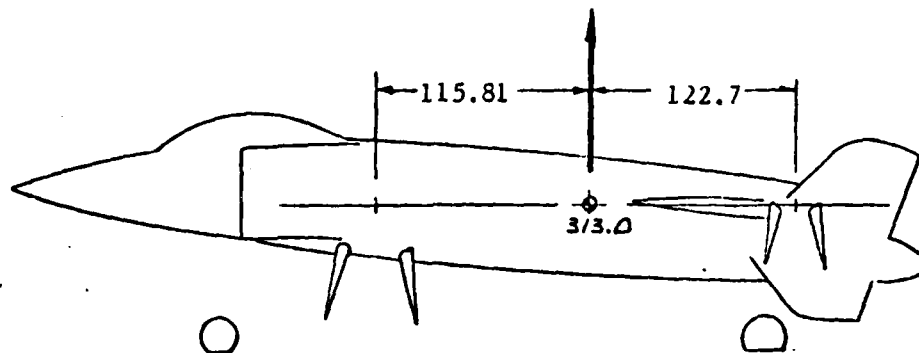
Figure 3-3 Geometric Relationships of the Wing and Canard Thrust Vectors

wing $\emptyset$ versus canard $\emptyset$ is shown in Figure 3-4. For each condition of
maximum trimmed lift and maximum trimmed lift with simultaneous control,
the numerical relationships of the canard and wing performance can be
evaluated. A wing $\emptyset$ of 1.60 is seen to be required to support a canard $\emptyset$
of 1.40 and provide full simultaneous control about all three axes during
hover.

Historically, data from previous tests indicate a 0.05 decrement of $\emptyset$ when
converting a rectangular model to a tapered and swept model provided the
taper ratio was 0.60 or greater. This 0.05 delta $\emptyset$ was added to the full
scale hardware requirement of $\emptyset_{wing}$ = 1.60 (assumes no scale or
temperature effects) to establish the 0.20 scale rectangular model
requirement of $\emptyset_{wing}$ = 1.65.

This increase in the wing performance requirement from $\emptyset_{wing}$ = 1.50
(full scale tapered hardware) eliminated Phase I of the development
program. Phase II was reoriented toward developing the higher performance
augmenter configuration. This redirected effort was organized in five
separate tasks.

The first task was to improve the Coanda/flap isolated performance through
the use of 0.20 scale rectangular models, full scale segment models, and
the full scale Coanda/flap article. Coanda/flap surface development was
concentrated on the wing aft augmenter flap (elevon) since this surface
was common to any tapered throat configuration (see Section 4.0). This
hardware development is presented in Sections 6.0 and 9.0.

The second task was to improve the centerbody nozzle configuration through
the use of an analytical computer code, 0.20 scale rectangular models,
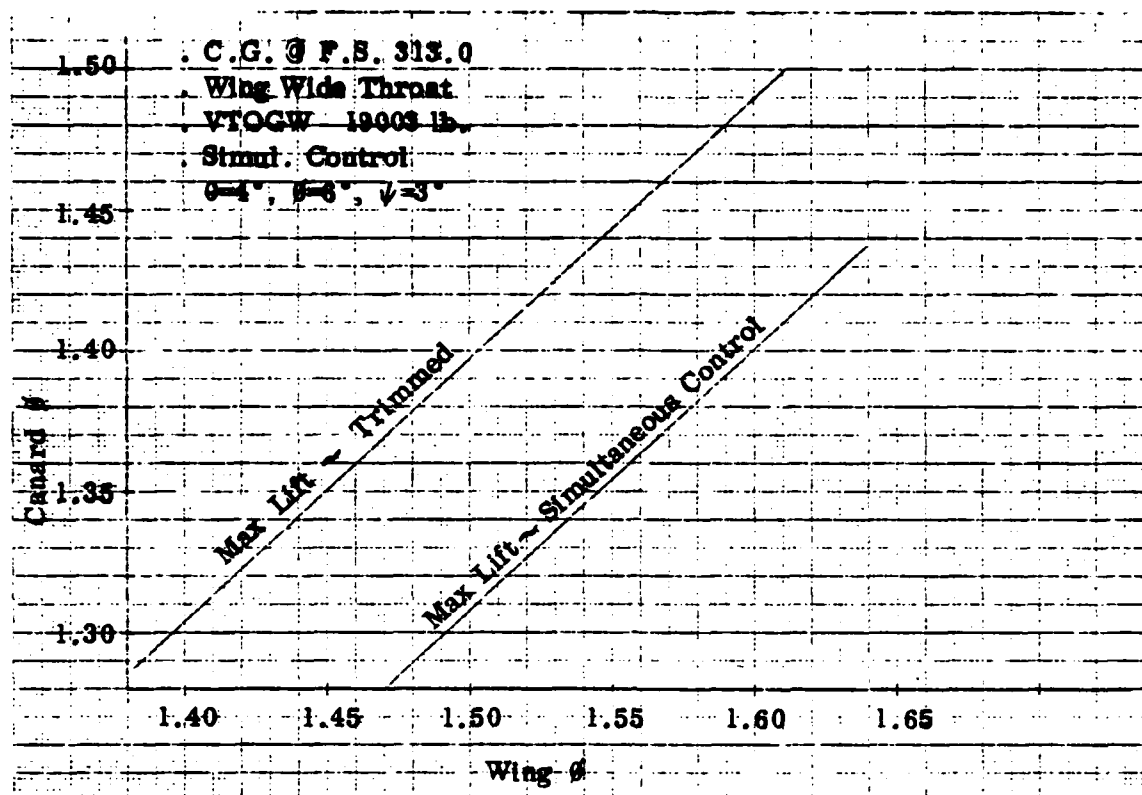
Figure 3-4   Wing ϕ Versus Canard ϕ

reduced scale segment models, and the full scale centerbody article. The analytical and 0.20 scale rectangular development is presented in Section 7.0.

The third task involved using the improved Coanda/flap and centerbody surfaces to determine the optimum performance configuration for the augmenter (inlet area ratio, diffuser flap length, end wall treatment, and inlet configuration) that could be integrated into the XFV-12A wing airfoil. Because the augmenter configuration is highly dependent on the centerbody nozzle configuration, the second and third tasks were conducted simultaneously and the development results are presented in Section 7.0.

The fourth task was to configure a 0.20 scale tapered model of the full scale configuration using the 0.20 scale rectangular model as the mid-span section cut of the tapered model.

The fifth task was to design and fabricate flight-weight full scale hardware for a full scale, hot, demonstration of the augmenter configuration.

3.2   SUMMARY

In an effort to improve the performance of the XFV-12A thrust augmenters, a two-phase remedial development program was initiated. The primary objective of Phase I was to correct deficiencies of the aircraft's augmenter components and demonstrate the original goal performance levels of ϕ_{wing} = 1.50 and ϕ_{canard} = 1.40. The objective of Phase II was to develop an improved centerbody nozzle along with necessary modifications to the augmenter shroud to increase augmenter performance. Initially, no specific performance goals were stated, however, subsequent operational studies indicated augmentation levels of ϕ_{wing} = 1.60 and ϕ_{canard} = 1.40 were necessary to assure adequate control response for the aircraft in high gust level conditions.

Due to this increased goal performance level, the Phase I development effort was terminated and Phase II was directed at achieving a ϕ_{wing} = 1.65 on the scale model development augmenter. The Phase II program was then composed of five basic tasks.

1. Develop Coanda flap components compatible with the higher goal augmentation ratios.

2. Develop new centerbody nozzles to maximize augmenter performance.

3. Optimize augmenter geometry with new Coandas and centerbody nozzles.

4. Design, fabricate, and test a .2 scale augmenter model representative of the full scale aircraft augmenter design.

5. Fabricate and test full scale L/H wing flight hardware and demonstrate ϕ_{wing} = 1.60.

To date, tasks 1, 2, and 3 have been initiated and their current status is discussed in the remainder of this report. Unfortunately, the contract effort was terminated before all tasks in the development program could be successfully completed.

4.0 DESIGN STUDIES SUMMARY

4.1   INTRODUCTION

Diagnostic tests, conducted on the XFV-12A Prototype #1 wing augmenter
hardware following the hover tests, identified various flow deficiencies
on the augmenter components. These diagnostic tests concluded that
component modifications to improve the flow qualities would be necessary
to obtain the performance previously achieved on the original full scale
hardware tested on the whirl rig. Design studies were utilized to
identify and control the modifications made to each component to assure
compatibility within the aircraft moldline constraints. These studies
were used to evaluate configurations, fabrication methods, weight, and
cost. To assure that component improvements did not create unacceptable
changes in the augmenter parameters, a mid-span cross-section cut of the
tapered wing augmenter was utilized to control and/or maintain the
augmenter parameters. This augmenter cross-section was also used as the
representative cross-section to establish the 0.20 scale augmenter model
utilized in the experimental test program.

Concurrent with the effort to remedy the flow quality problems, a desire
for increased performance was imposed upon each component.

This section will address the design requirements for the Coanda shape,
centerbody nozzle, and the configuration evolution studies.

4.2   COANDA SHAPE DEVELOPMENT

Data from the isolated tests of Reference (6) on the aft flap (wing
elevon) demonstrated highly outboard angular flow in the inboard region in
addition to poor nozzle total pressure distribution. A comparison of the
elevon surfaces from the XFV-12A Prototype #1 and the original full scale
whirl rig hardware reveals a large difference between the feed duct
location relative to the nozzle exit, Figure 4-1. This nozzle/feed duct
configuration from the XFV-12A elevon did not allow adequate distance for
the flow to turn effectively and, therefore, exhibited outboard angularity
as demonstrated during the isolated tests of the wing elevon.

Derivation of a Coanda shape has until recently been challenging because
of the limited airfoil thickness available in a supersonic airfoil. The
requirements of large initial radius (R_o) and maximum running length to
the augmenter throat (s_{throat}) were not compatible with supersonic
airfoil thicknesses. Articulated concepts were utilized on the XFV-12A
canard; however, the penalties of weight, fabrication requirements, and
operation under load presented valid arguments for alternate concepts. A
convenient shape which was easily definable mathematically and allows a
large R_o at the nozzle exit with a gradual increasing curvature is the
log spiral.

The mathematical equations for a log spiral shape are shown in Figure 4-2.

After reserving space for the internal nozzle shaping (dimension X), the
remaining width (dimension Y), shown in Figure 4-2, is available for the
external shape of the Coanda. Since a number of shapes with various

4-1

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Original Full Scale
Whirl-Rig Elevon

XFV-12A Proto #1
Elevon

XFV-12A Proto #1
Elevon Feed Duct

Original Full Scale
Whirl Rig Elevon
Feed Duct
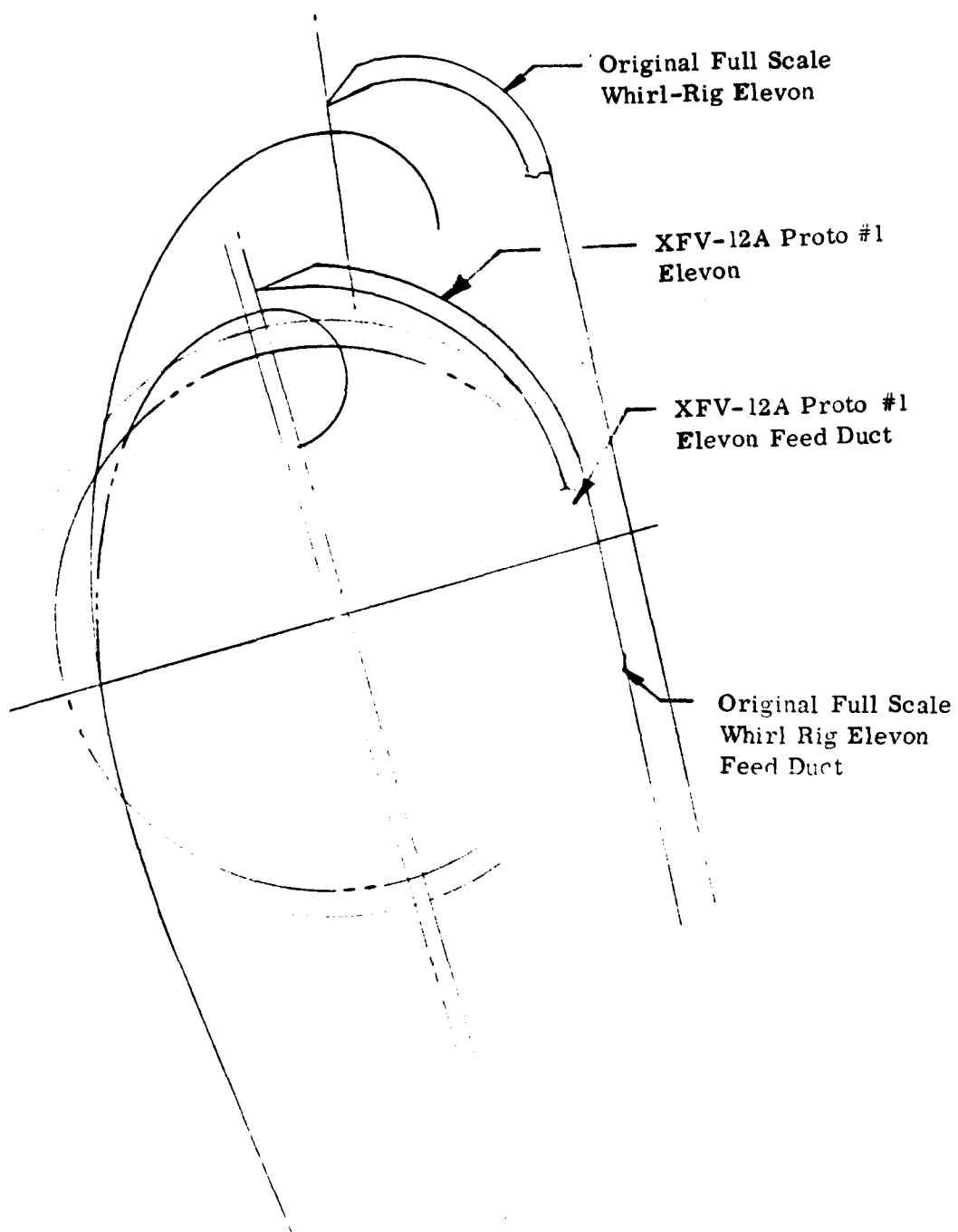
Figure 4-1   Comparison of XFV-12A Proto #1 Elevon and Original Full Scale
Whirl Rig Elevon Nozzle Shapes
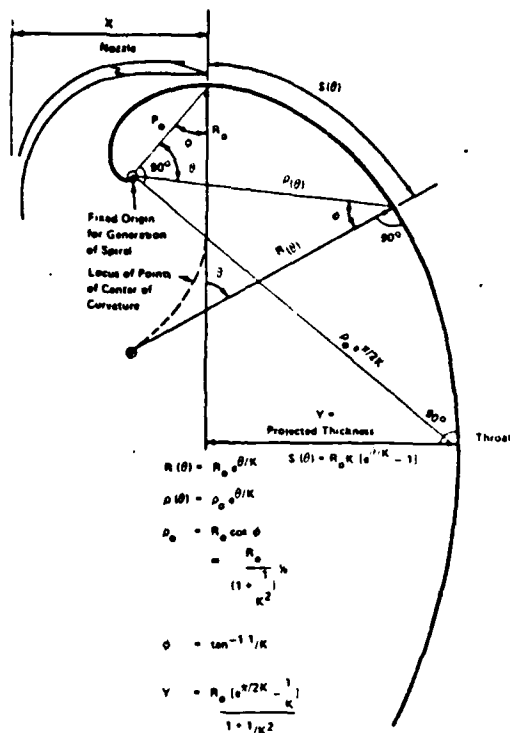
Figure 4-2 Spiral Definition

combinations of spiral constant (K) and initial radius (R_o) can be configured as shown in Figure 4-3, a method of determining the optimum shape was required.
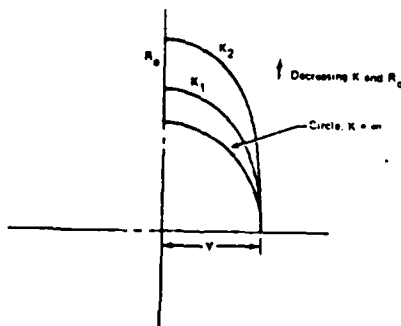
Figure 4-3 Example of Various Spirals with Identical Thickness

From experimental separation criteria, universal jet growth laws, and the geometric relationships of initial radius (R_O), running length to the throat (S_{throat}), nozzle gap (t), and projected Coanda width (Y) (Figures 4-4 through 4-6), a carpet plot of jet growth, spiral K factor,
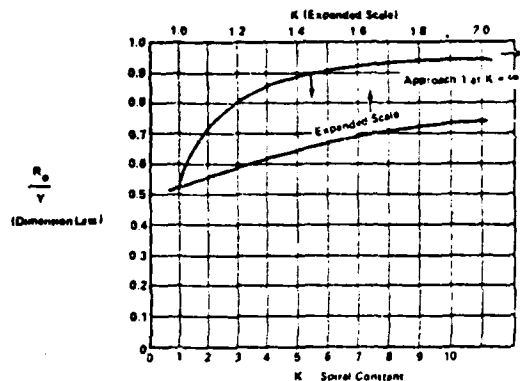
Figure 4-4   Initial Radius R_O Versus K
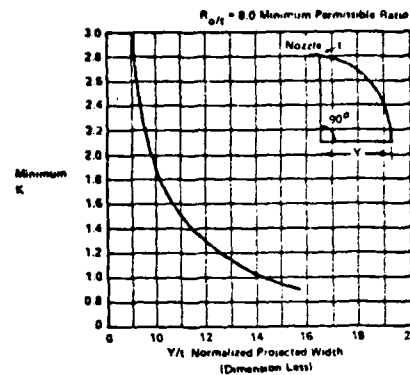
Figure 4-5   Minimum K Versus Y/t
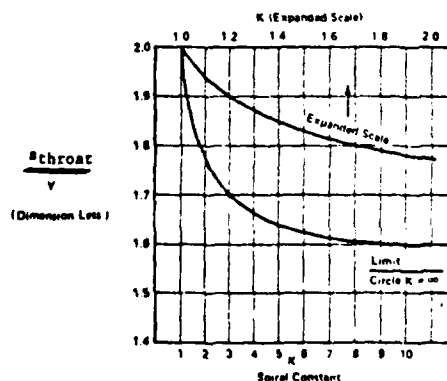
Figure 4-6   Running Length from Nozzle to Throat Versus K

and projected width is shown in Figure 4-7. The boxed areas are the packaging regimes for the XFV-12A wing and canard augmenters utilizing the moldline constraints of the existing airfoils.

As shown in Figure 4-7, the spiral constant (K) for the wing Coandas could vary from a K = 1.17 to a K = 1.50. To determine the optimum K factor, an experimental test program was developed to quantify performance increments of the revised shape and, as noted earlier, resolve the flow angularity problem of the XFV-12A elevon. This test program concluded that a log spiral shape with a K factor of 1.30 did improve the isolated performance, and simultane- ously relocated the nozzle further from the feed duct as shown in Figure 4-8. For details of this test program see Section 6.0. Figure 4-8 compares the improved aft flap nozzle shape and the XFV-12A Proto #1 aft flap nozzle shape.

4.3   CENTERBODY DEVELOPMENT

Data from the isolated test of Reference (6) on the XFV-12A Prototype #1 centerbody demonstrated reduced chordwise spreading when installed in the augmenter, but isolated tests of the same centerbody nozzle showed normal spreading characteristics. A comparison of the centerbody nozzle shapes from the original full scale whirl rig hardware and the XFV-12A prototype nozzle, Figure 4-9, reveals a more blunt base on the XFV-12A prototype centerbody due to the addition of an upper surface close-out door. Hypotheses of probable causes of the reduced spreading included the influence of the poor end flow conditions existing on the prototype augmenter and possible separation off the centerbody's blunt base. Since the XFV-12A Prototype centerbody was also installed at a lower position in the augmenter than the original full scale whirl rig hardware, the influence of the Coanda jets could have contributed to the reduced spreading. Resolution of the above problems were to be addressed in the development of an improved centerbody nozzle configuration.

Three types of centerbody nozzles were investigated for packaging requirements and are identified as; hypermixing, asymmetric and symmetric.

4.3.1   Hypermixing Nozzle

The hypermixing nozzle features alternating jets that create streamwise vortices, Figure 4-10, which accelerate the turbulent mixing and thus increase the entrainment. The amount of hypermixing angle was evaluated experimentally and analytically to optimize the performance potential in the wing augmenter. Results of this experimental and analytical program are presented in Section 7.0. Packaging restraints did not restrict this investigation as shown in Figure 4-11. Figure 4-11 presents a typical cross-section of the centerbody nozzle showing the existing centerbody hypermixing nozzle of the XFV-12A and the highest hypermixing angle possible without affecting the upper panel closeout door.

Because of the limited potential performance of this nozzle configuration, development of this concept was terminated in favor of configurations with higher performance potential.
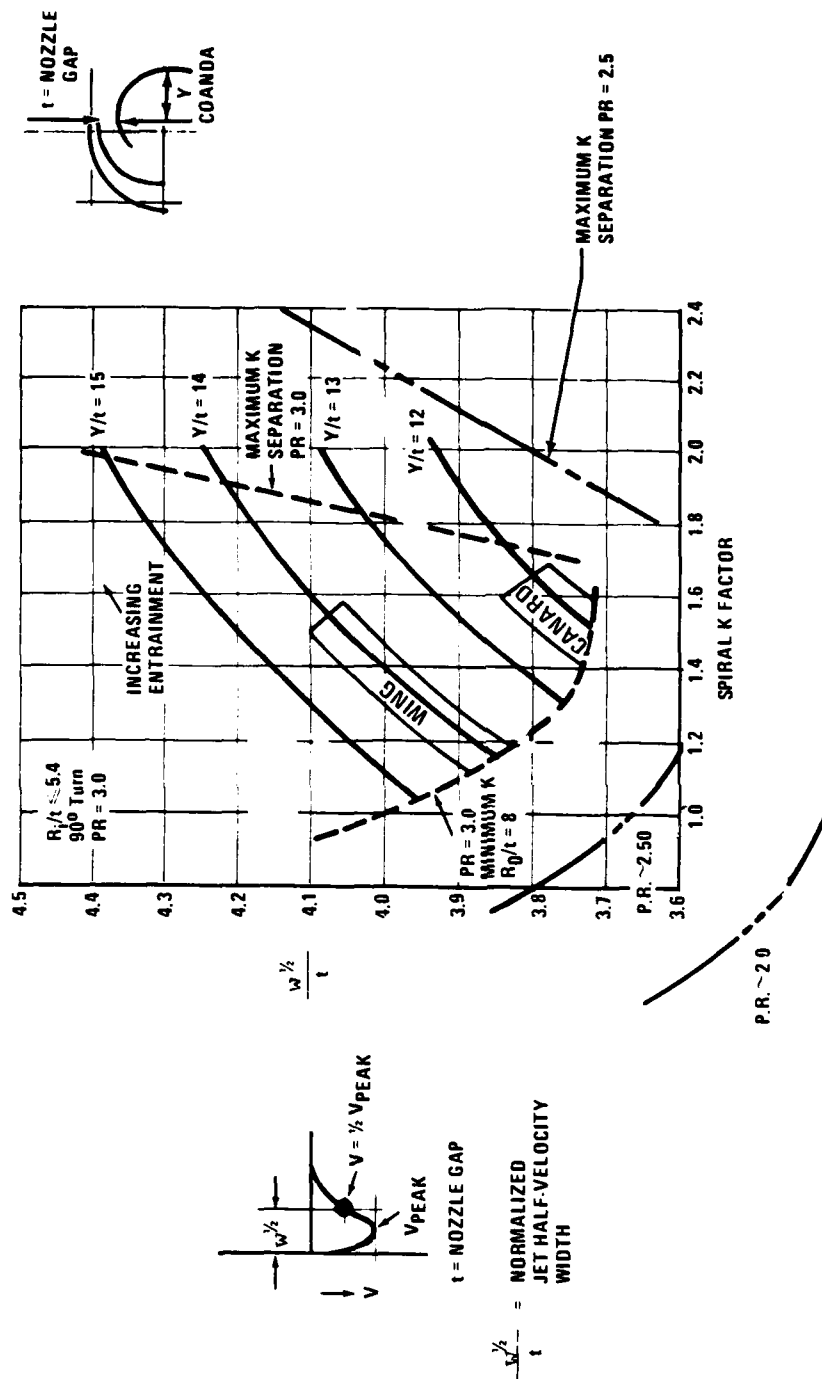
Figure 4-7 Empirical Coanda Selection Criteria

XFV-12A Proto #1 Elevon

Improved Elevon
Nozzle Shape
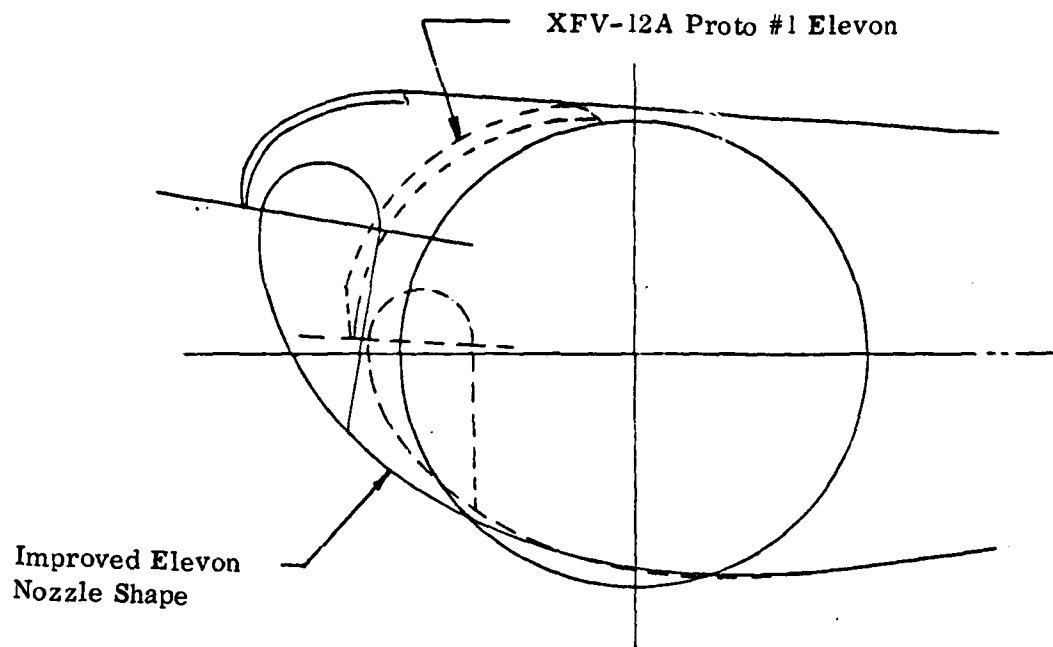
Figure 4-8   Comparison of Coanda Shapes

Original Full-Scale
Whirl Rig Centerbody
Nozzle

XFV-12A Proto #1
Centerbody Nozzle
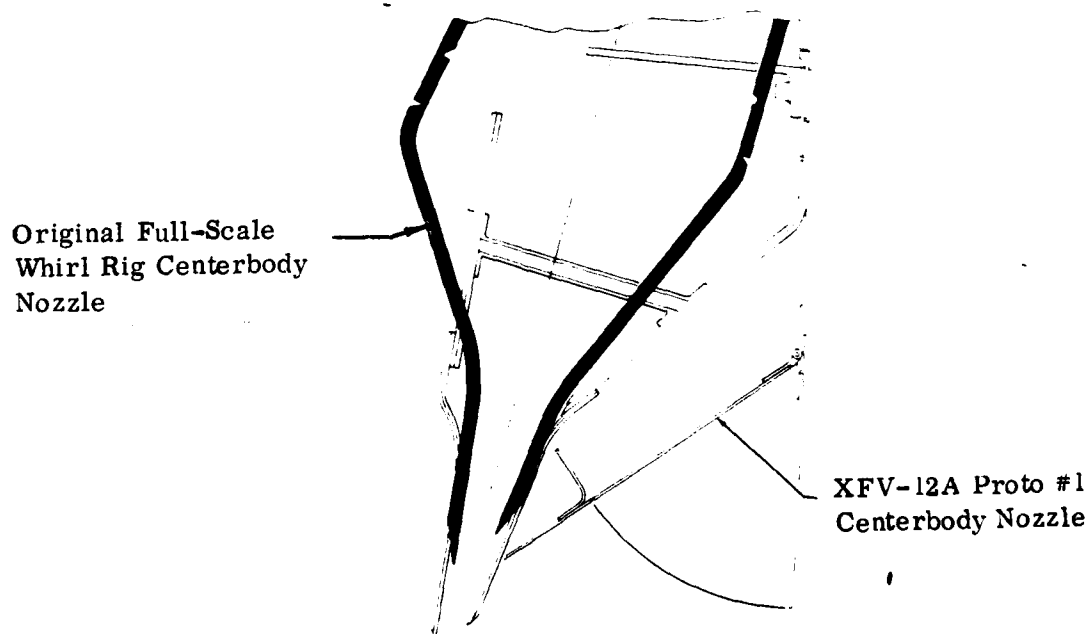
Figure 4-9   Centerbody Shape Comparison
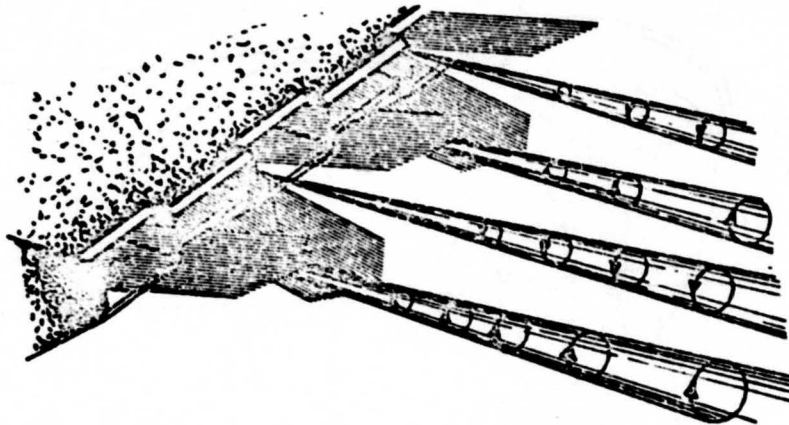
Figure 4-10 Hypermixing Nozzle
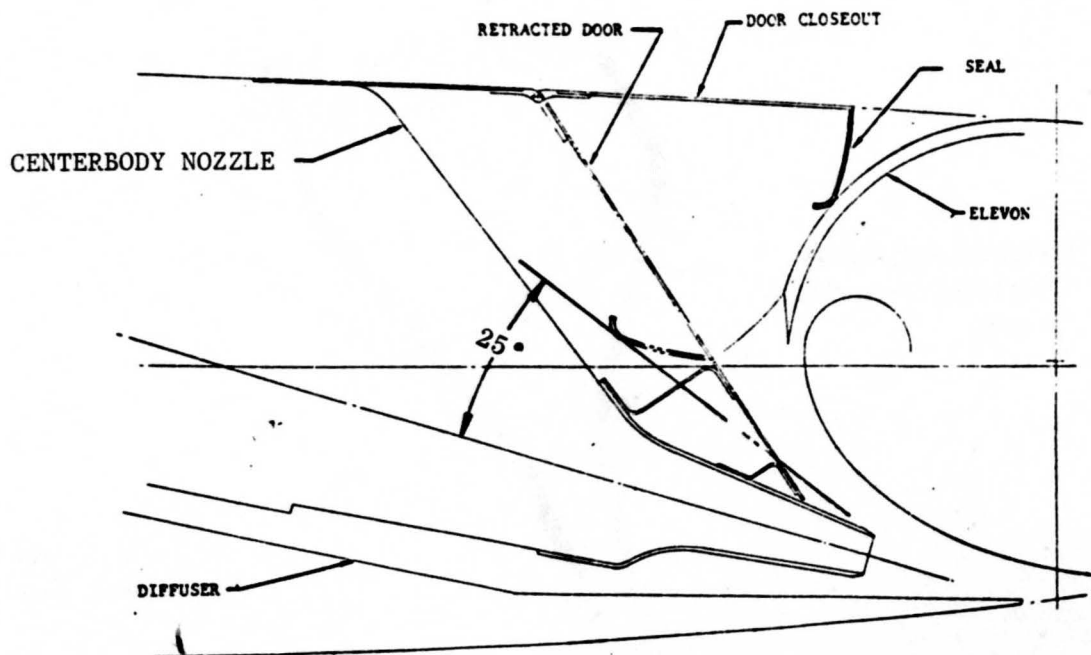
Figure 4-11 Typical Cross-Section of the Centerbody Nozzle

4.3.2   Asymmetric Nozzle

Previous experimental testing had identified a centerbody nozzle configuration that combined a series of aft facing spanwise slots with alternating series of cross-slots on the forward side, shown in Figure 4-12, that would produce increased mixing capability. This concept was advantageous from a packaging viewpoint since it provides its own closeout system on the upper surface as shown in Figure 4-13.
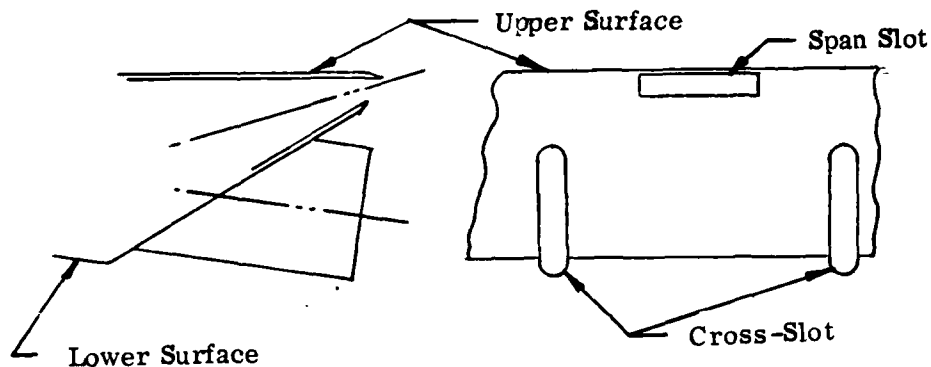
Figure 4-12   Asymmetric Centerbody Nozzle
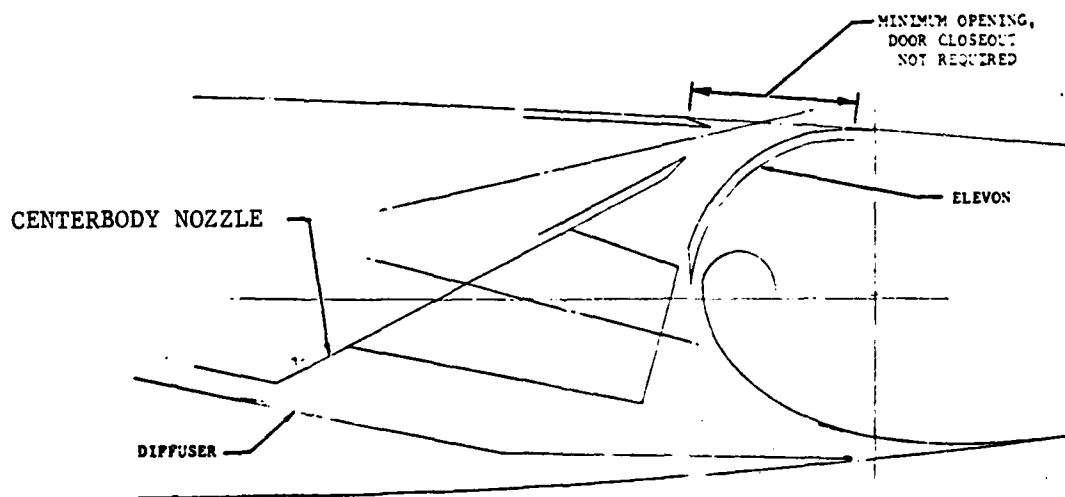
Figure 4-13   Closeout System on the Upper Surface

Packaging constraints only dictated the total width as shown in Figure 4-13 and did not impair the investigation into the number of cross-slots, flow split between the span slots versus the cross-slots, or the launch angle of the nozzles. When the initial performance values from the experimental test model utilizing this nozzle configuration were lower than expected, this nozzle configuration and the development tasks associated with this configuration were terminated prior to completion. The experimental and analytical results of this nozzle configuration are presented in Section 7.0.

4.3.3 Symmetric Nozzle

The symmetric multi-lobed nozzle, shown in Figure 4-14, consists of a series of symmetric cross slots equally spaced along the span. This
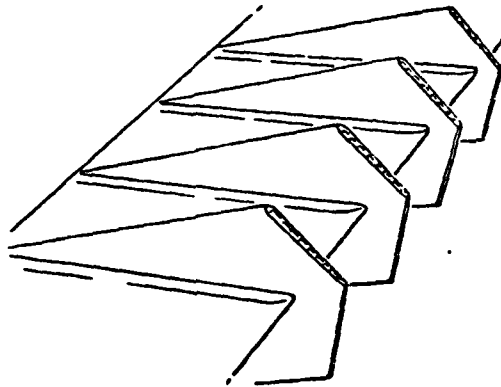
Figure 4-14 Symmetric Centerbody Nozzle

configuration provides increased chordwise jet launching thereby increasing the turbulent mixing capability. The primary disadvantage to this centerbody nozzle configuration is the unsuccessful effort to date to provide a closeout system on the upper surface of the wing airfoil.

Packaging considerations were restricted by the overall width of the nozzle, but being symmetric around the nozzle centerline the realistic restraint is the proximity of the forward flap surface. The nozzle configuration could vary between the boundaries shown in Figure 4-15 with various combinations of nozzle parameters. These parameters are identified by Figure 4-16. All of the parameters are independent of each other except the wedge and launch angle and must maintain the relationship as shown in Figure 4-17.

Since the program was terminated by funding constraints prior to completion, only the recommended nozzle configuration (based upon experimental and analytical trends) can be presented. This nozzle
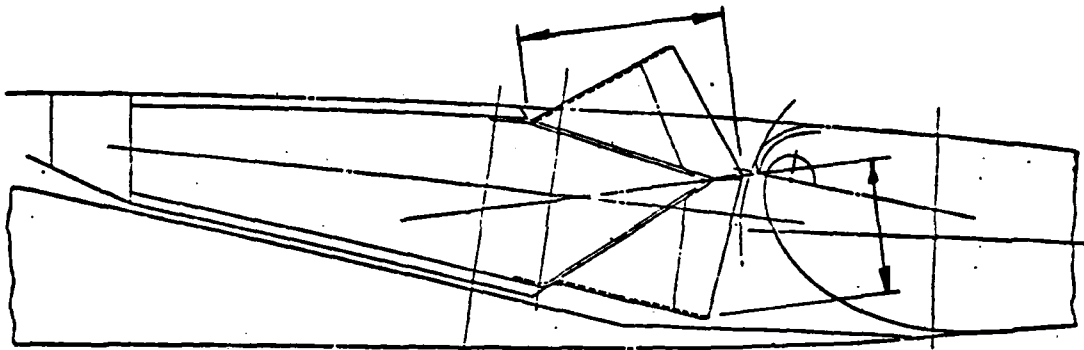
Figure 4-15   Nozzle Configuration
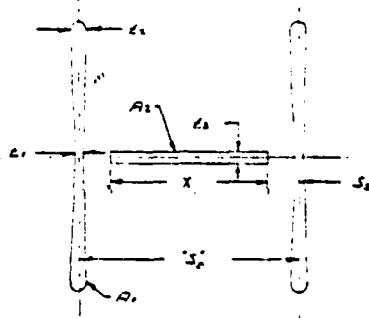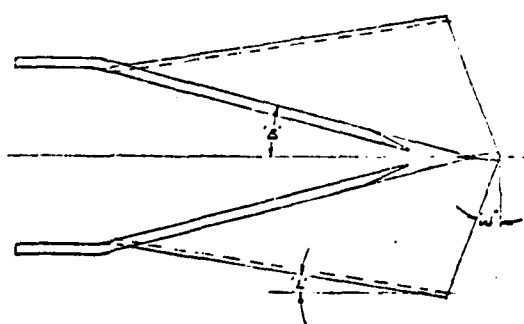
Figure 4-16   Nozzle Parameters

4-11
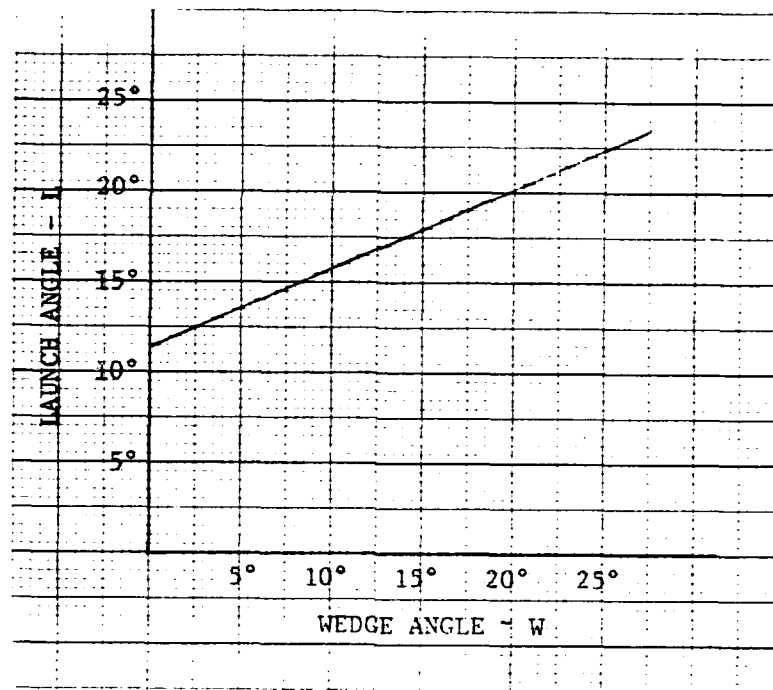
Figure 4-17 Wedge Versus Launch Angle

configuration contains a 25° base angle, 21° launch angle, with a
linerarly varing wedge angle, 15 element cruciform with a two to one
bow-tie ratio on the cross-slots. Figure 4-18 presents the recommended
nozzle configuration packaged in the wing cross-section.

4.4   CONFIGURATION EVOLUTION

4.4.1   Wide Throat

Experimental testing of a 0.20 scale rectangular model in the laboratory
indicated that a throat width increase (inlet area ratio A_2/A_0) was
required in the wing augmenter to fully utilize the higher entraining
potential of either the asymmetric or symmetric centerbody nozzles. The
XFV-12A prototype hardware had the flexibility initially incorporated in
the wing for a throat width increase by revising the forward flap hinge
line and modifying the centerbody rotation as shown in Figure 4-19. The
narrow throat (XFV-12A prototype hardware) or the wide throat utilizes
identical surfaces and the flexibility is accomplished by hinge line
location *relative to the surfaces.* Because of this design flexibility,
packaging of increased throat width could be combined with improved
Coanda/flap surfaces and an improved centerbody nozzle. Figure 4-20 shows
a planform and three section cuts of the wing wide throat. The mid-span
cross-section cut is the control section mentioned earlier. This
configuration established the model configurations for experimental and
analytical studies. If the tapered and swept augmenter studies had
continued, this configuration would have established the control lines for
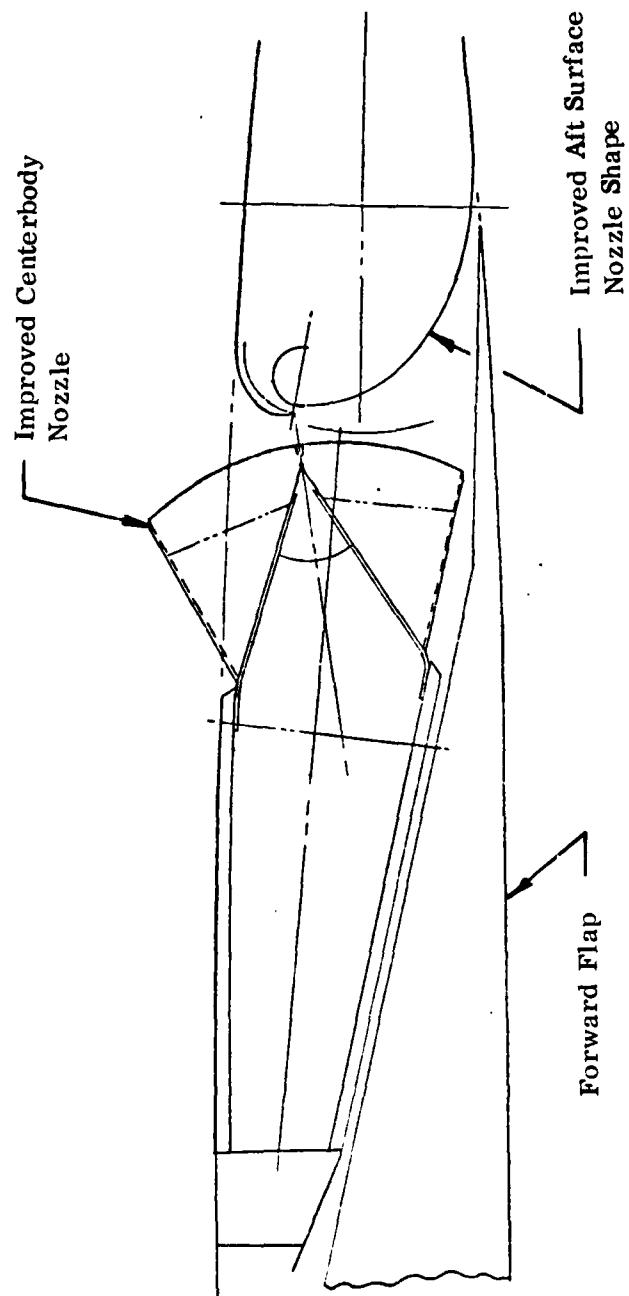
Figure 4-18. Wing Cross-Section with Improved Centerbody Nozzle

Improved Centerbody Nozzle

Improved Aft Surface Nozzle Shape

Forward Flap

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INCREASED CENTERBODY
ROTATION ~ WIDE THROAT

FORWARD FLAP ~ NARROW THROAT

WIDE THROAT H.L.

NARROW THROAT H.L.

FORWARD FLAP ~ WIDE
THROAT
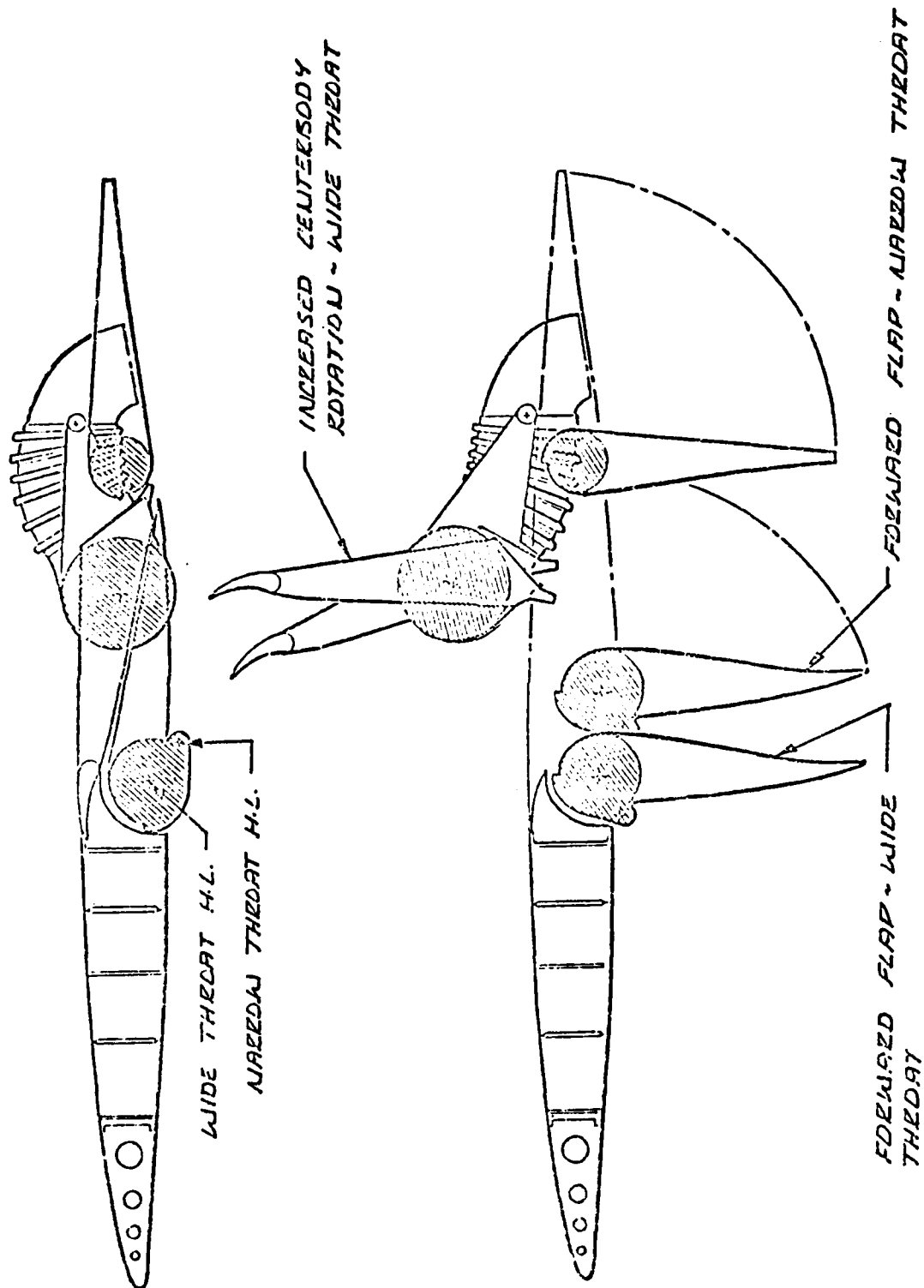
Figure 4-19   Flexibility in Wing Augmenter
Design

EJECTOR PRIMARY HINGE

ELEVON HINGE

$Y_{wop} \cong 0$

TR $Y_{wop} \cong 0$
$Y_{wop} \cong 0$

TR ℓ_{wd}

TR $Z_{ball} \cong 0$

TR DIFF HINGE

$Z_{wop} \cong 0$, $Z_{wop} \cong 0$, $Z_{wop} \cong 0$

TR $Y_{wop} \cong 0$
$Y_{wop} \cong 0$

TR $\ell_{ml} \cong 0$

DIFFUSER HINGE
BAL 1

CUT "13", $Y_{wop} \cong 85.8321 = X_{wop} \cong 0$
VIEW ROTATED INTO PLANE OF PAPER

$Y_{wop} \cong 0$

TR ℓ_{ml}

ℓ_{wd}

TR DIFF HINGE

& THROAT

ELEVON HINGE

MID-SPAN · $X_{wop} = 37.8963$
VIEW ROTATED INTO PLANE OF PAPER

ELEVON HINGE

EJECTOR PRIMARY HINGE

PRIMARY EJECTOR HINGE

ELEVON BEARING
X_{WEL} 34.625	X_W 33.092
Y_{WEL} 0.00	Y_W 104.3192
Z_{WEL} 0.00	Z_W 0.109

ELEV NOZZLE
X_{WEL} 36.325

TR X_W 0

74° TRUE

88.5° TRUE

−16.68136301°
(116° 40.8817')

$\mathcal{C}$ THROAT (116° 40.8817')

END OF THROAT (TRUE)

B 250 ID

SECONDARY EJECTOR BEARING (STOWED)

TR X_{WDF} 4175

END OF DIFF NOZ ROTATED
X_{WDF} 45.00	X_W 40.456
Y_{WDF} 48	Y_W 122.551
Z_{WDF} .326	Z_W 2.996

CUT "A" — X_{WDF} 0
VIEW ROTATED INTO PLANE OF PAPER

EJECTOR PRIMARY HINGE (WEP)

ELEVON HINGE

X_{WDF} 2 0 Z_{WDF} 0, Z_{WDF} 0

B 00 ID

YELLOW PLANE

DIFF BEARING
X_{WDF} 34.438	X_W 30.8150
Y_{WDF} 0.000	Y_W 118.3568
Z_{WDF} 0.000	Z_W 3.5173

Y_W 90.5114

YELLOW PLANE

X_W 0.0000

356-918509

2

WING THROAT TAPER DIAGRAM
(LOOKING NORMAL TO WING REF PLANE)
NO SCALE

PARAMETER   WIDE THROAT DESIGN

THROAT GAUGE (Dr)   34.00 IN
THROAT @ AUG T (Dt)   20.10 IN
TAPER RATIO AUG (λ)   .609
SPAN AUG (b)   108.80 IN
THROAT A AUG (A₂)   2707 IN²
NOZZLE EXIT A (A₀)
A₂/A₀ AUG
EXIT A AUG (A₃)   4560 IN²
H₃/D AUG
THROAT EXIT AUG (L)
L/D AUG
(DA/DA AUG
     ELEVON   R₀ = 3.285 - 1.444
               R₁ = 1.05 - 52
               K = 1.3
      FWD DIFF   Z₀ = 3.008 - 1.223
               R₁ = 1.060 - 539
               K = 1.3
ASPECT RATIO $\frac{SPAN}{A_c}$ 4.37
A₀ DISTRIBUTION AC   DIFF
               CB
               ELEV
               EWB R
               EWB T

NOZZLE SPAN AT   DIFF   09.39 IN
               CB   13.639 IN
               ELEV   05.650 IN
               EWB R
               EWB T
C'BODY ABOVE T₊T (w)
FEED DIST (b₀-CB EL)   21.4% - 57.2%, 21.4%.

FIGURE   4-20   WIDE THROAT PLANFORM

NOTES   UNLESS   OTHERWISE   NOTED

356-918509

				HEAT TREAT				FOR PARTS LIST SEE PL	
					UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES	CONTR NO		Rockwell International Corporation Columbus Aircraft Division	
					MFG TOLERANCES ON UNDIMENSIONED : .03	DWN BY C. BALCER			
				FINISH	X (DECIMALS) : .03 XXX (DECIMALS) : .010 ANGLES : 30	MF BY		WING DESIGN ENVELOPE & PARAMETERS - WIDE THROAT 1 FV-12 A (DESIGN LAYOUT)	
					HOLES NOTED DRILL .013 THRU .040 : .001 - .001 .041 THRU .130 : .002 - .001 .131 THRU .270 : .003 - .001 .330 THRU .500 : .005 - .001 .501 THRU .750 : .005 - .001 .771 THRU 1.000 : .007 - .001 1.001 THRU 2.000 : .010 - .001	APPD			
ITEM QTY	NEXT ASSY	USED ON	NO ITEM NO THRU				SIZE CODE IDENT NO J 89372	356-918509	
PROD PER END ITEM	APPLICATION	EFFECTIVITY			DO NOT SCALE PRINT		SCALE	SHEET	

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hardware development. A comparison of the wing augmenter parameters for the XFV-12A prototype hardware and the wide throat configuration are shown in Table 4-1. Because the same surfaces are utilized with either the wide throat configuration or the XFV-12A prototype hardware, an increase in the throat width produces a decrease in the augmenter parameter L/W.

Table 4-1   Wing Augmenter Parameters

PARAMETER	WIDE THROAT	XFV-12A PROTO 1
A_2/A_0	22.304	15.488
A_0 Total	121.365 In^2	121,911 In^2
A_0 Distribution	16.6-58.3-19.4	21.3-48.6-23.1
A_2	2702 In^2	1888.33 In^2
Span	108.80 In.	101.62 In
A_2 Local	25.20 In	23.52 In.
L/W	1.25	1.95
R_0/t	Diff 10.39 Elev 10.46	Diff 11.58 Elev 8.08
Spiral K	1.30	-
Duct Area		
. Diffuser	50.27 In^2	50.27 In^2
. CB	140.6 In^2	140.6 In^2
. Elevon	50.64 In^2	50.64 In^2

4.4.2   Blended Wing Study

Concurrent with augmenter development, studies were being conducted on operational configurations and their mission performance evaluated. From these studies evolved a co-planar wing and canard configuration shown in Figure 4-21. This configuration contains an upper inlet and fuselage mounted landing gear as opposed to the side inlets and wing tip mounted landing gear of the XFV-12A. These features, along with the blended wing, result in a supersonic wave drag reduction and, therefore, increased mission performance. This blended wing configuration also permits the use of rectangular augmenters in lieu of swept and tapered augmenters typical of the XFV-12A configuration. This latter advantage allows increased vertical lift and performance. Swept and tapered augmenters investigated to date are lower in performance, as shown in Figure 4-22, because of spanwise flow problems associated with the tapered flap length.

A design study was initiated to investigate the feasibility of incorporating the blended wing concept on the XFV-12A prototype fuselage for a low speed and vertical hover demonstrator.

This configuration, designated the XFV-12B, would utilize hardware assets of the XFV-12A program to the maximum extent possible.
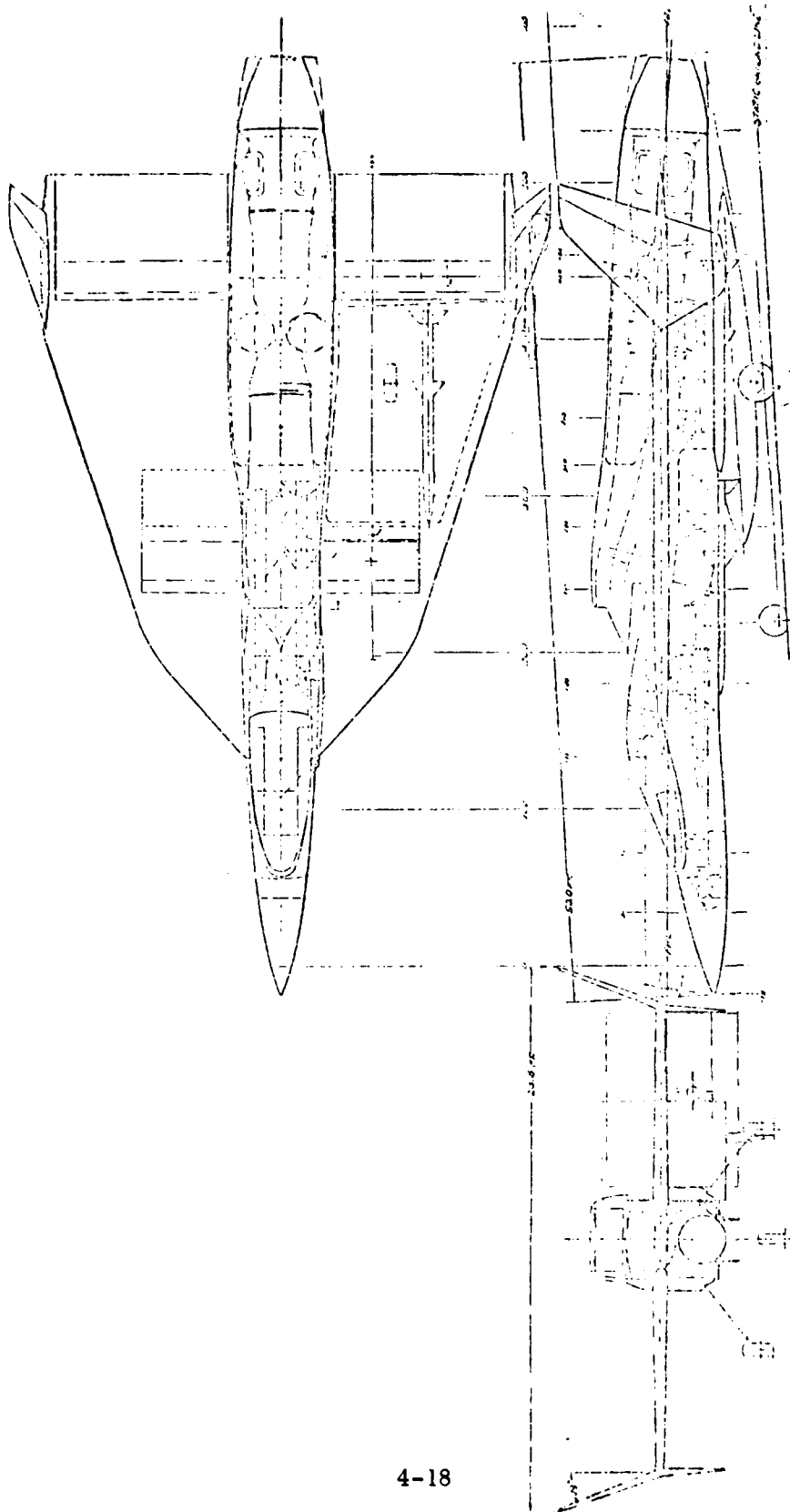
Figure 4-21. Co-Planar Wing and Canard Configuration
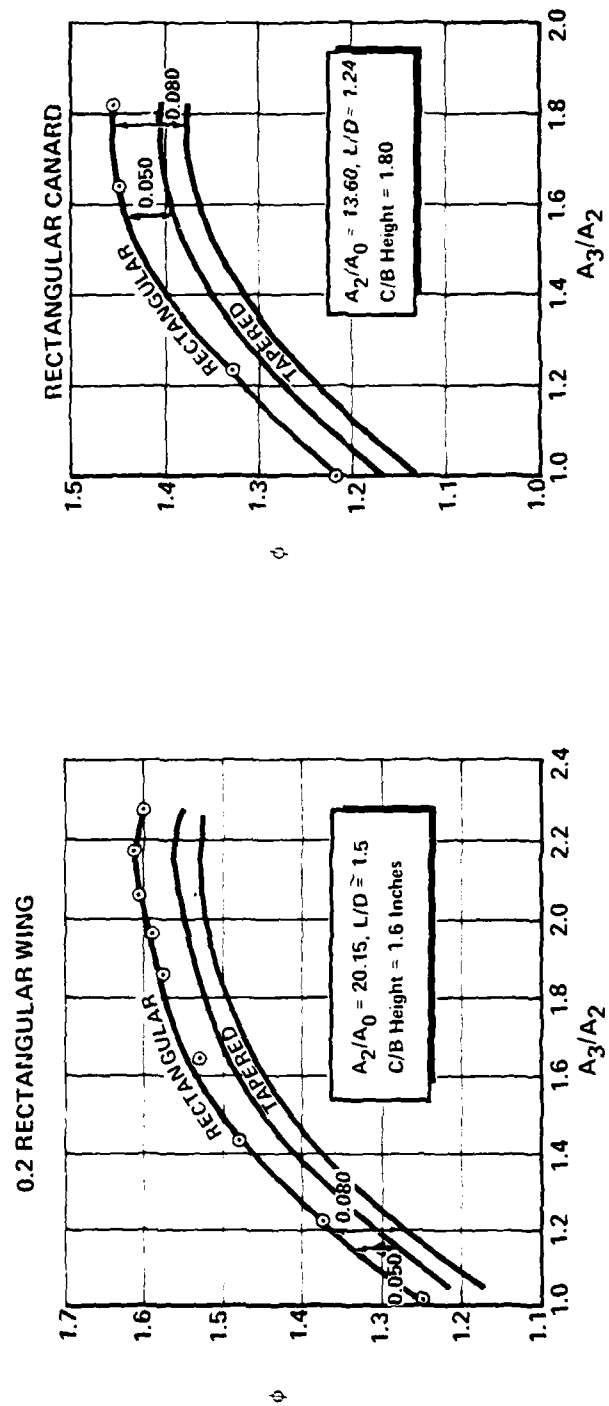
Figure 4-22. Augmenter Performance Tapered vs. Rectangular

RECTANGULAR CANARD

$A_2/A_0 = 13.60, L/D = 1.24$

C/B Height = 1.80

0.2 RECTANGULAR WING

$A_2/A_0 = 20.15, L/D = 1.5$

C/B Height = 1.6 Inches

4-19

Figure 4-23 Blended Wing Concept

COL 20933A

$V_{IAS} \sim 300$ Knots or $0.6M_N$ Whichever Is Less

Sink Speed ~ 15 Feet/Second Maximum

Symmetrical Pull-up ~ -1.0 to 3.0 G's

Rolling Pull-out ~ 1.0 to 2.4 G's with Maximum Roll Rate ~ 100°/Second

Gusts ~ 50 Feet/Second

Altitude ~ 0 to 25,000 Feet

50-Hour Maximum Engine Power

CTOL Flying Qualities Per MIL-F-8785

Transition and Conversion Flying Qualities Per MIL-F-83300

Augmenter Components

- 900°F
- 25 PSI Limit Internal
- 500-Hour Hot Life

COL 20934

Table 4-2. Design Criteria
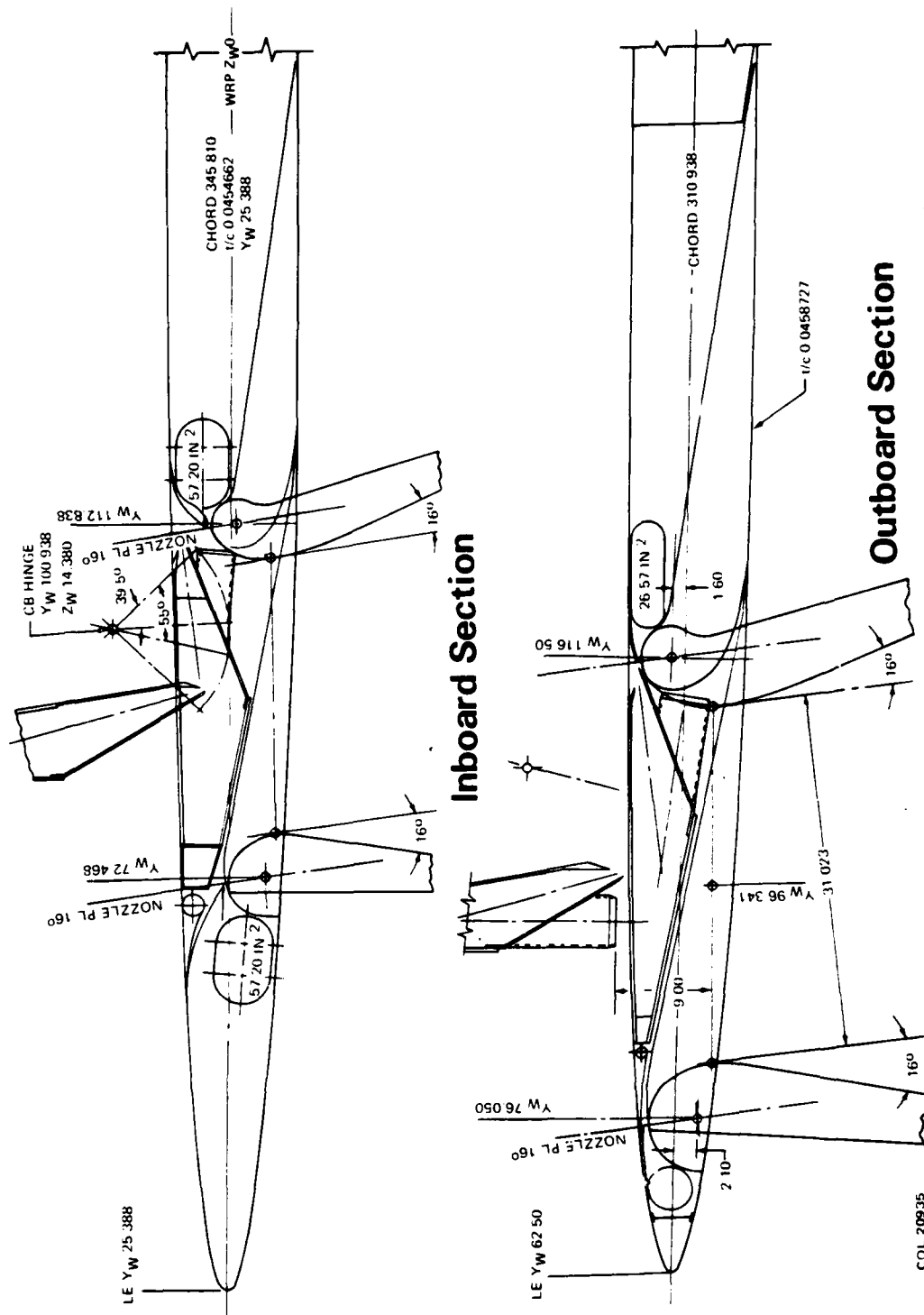
Figure 4-24. Forward Augmenter Cross-Sections
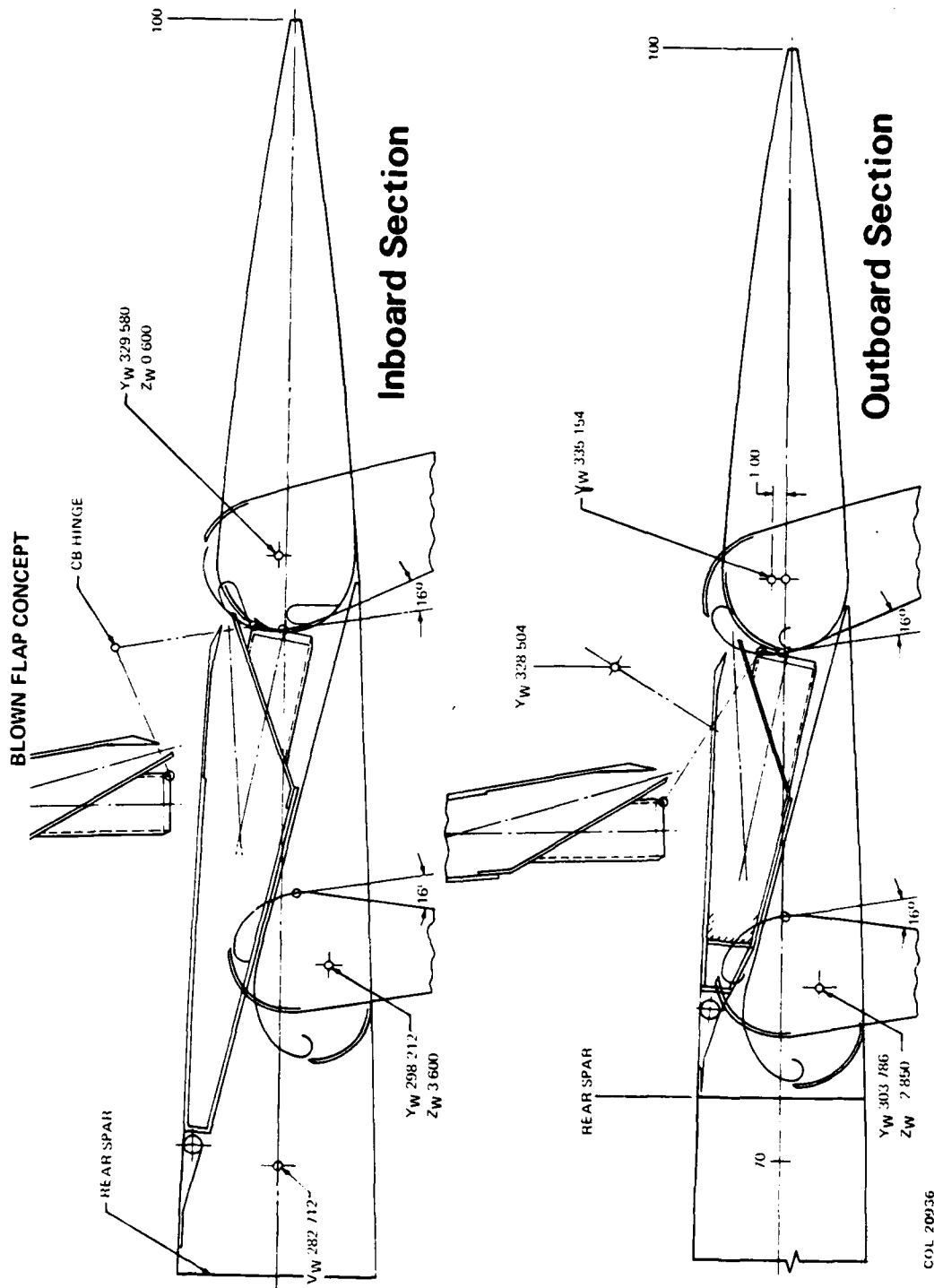
Figure 4-25.   Aft Augmenter Cross-Sections

PARAMETER	BLENDED	TPA 175
A₂/A₀	13.80	13.600
Ao TOTAL	145.0 IN.²	153.38 IN.²
Ao DISTRIBUTION	18-60-18	15.5-63-15.5
A₂	2001 IN.²	2086 IN.²
SPAN	64.50 IN.	70.00 IN.
A₂ LOCAL	31.023 IN.	29.80 IN.
L/D	1.512	1.240
Ro/t	8.00	8.00
SPIRAL-K	1.60	DIFF 1.80 ELEV 1.20
DUCT AREA:	--	--
DIFFUSER	52.20 IN.²	90.00 IN.²
C.B.	174.0 IN.²	165.0 IN.²
ELEVON	58.0 IN.²	63.62 IN.²

Forward Augmenter

PARAMETER	BLENDED	CONTRACT
A₂/A₀	20.50	22.304
Ao TOTAL	121.80 IN.²	121.37 IN.²
Ao DISTRIBUTION	18-60-18	16.6-58.3-19.4
A₂	2497 IN.²	2707 IN.²
SPAN	123.997 IN.	108.80 IN.
A₂ LOCAL	20.137 IN.	25.20 IN.
L/D	1.69	1.25
Ro/t	12.00	DIFF 10.39 ELEV 12.075
SPIRAL-K	1.30	1.30
DUCT AREA:	--	--
DIFFUSER	70.05 IN.²	50.27 IN.²
C.B.	146.16 IN.²	140.6 IN.²
ELEVON	70.05 IN.²	50.64 IN.²

Aft Augmenter

Table 4-3. XFV-12B Augmenter Parameters

Figure 4-26.   Aft Augmenter Cross-Section

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NR81H-50

Aft Augmenter Feed System

Forward Augmenter Feed System
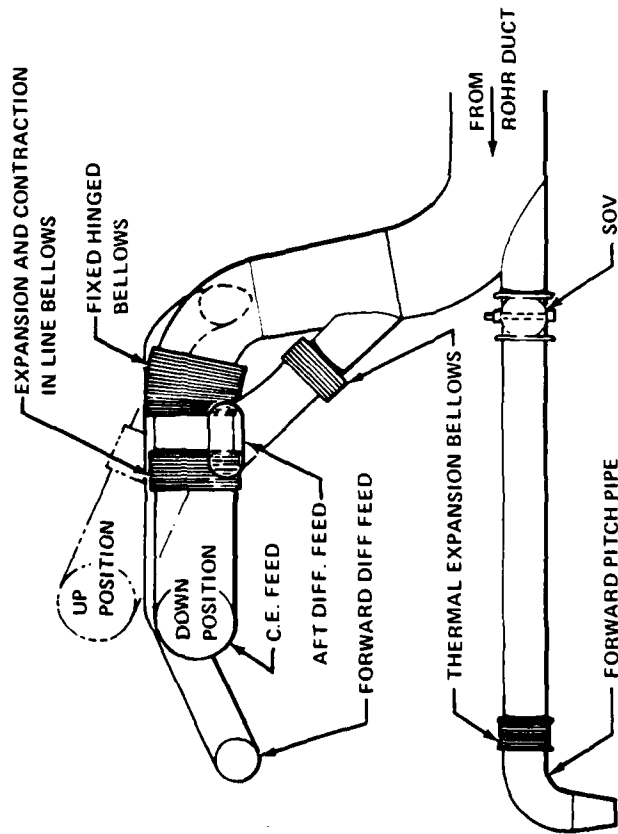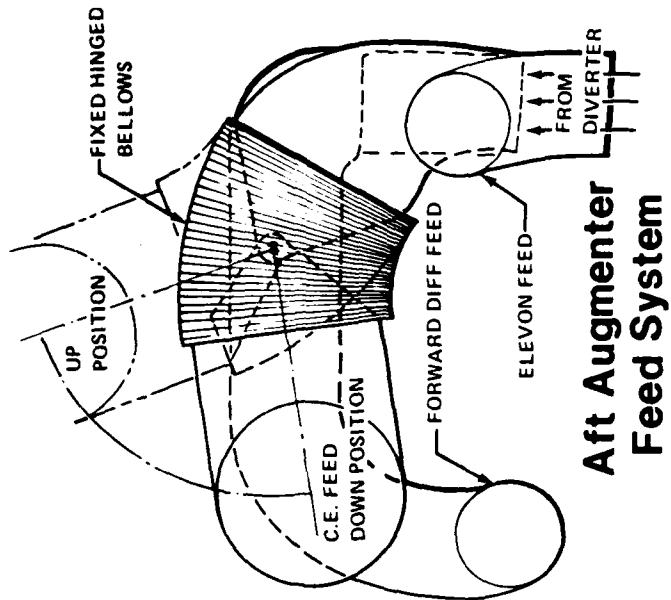
Figure 4-27 Ducting System

4-27

FORM 351-F REV 5-77
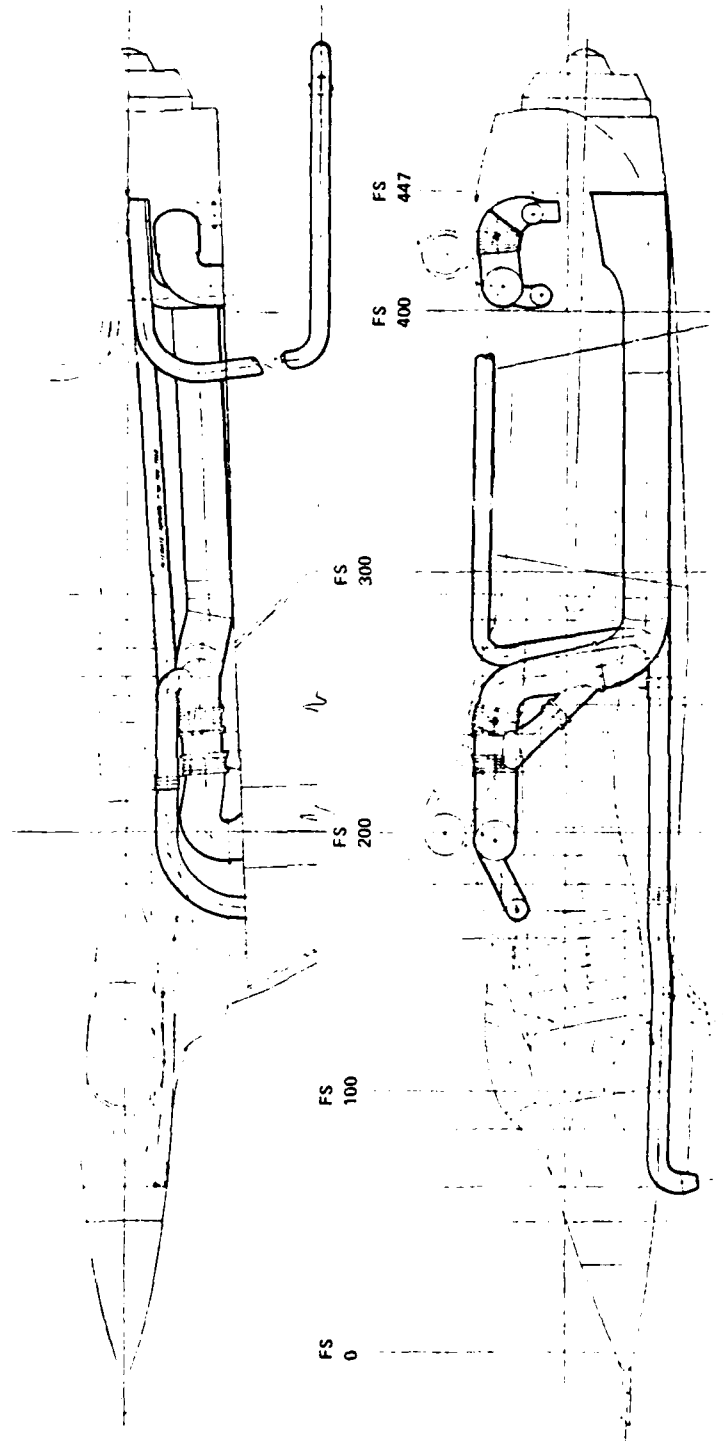
Figure 4-28 Ducting System

PITCH CONTROL PIPE

XFV-12A VERTICAL LOCATION

AFT AUGMENTER

MLG TRUNNION

INTEGRAL FUEL TANK

FORWARD AUGMENTER

CG
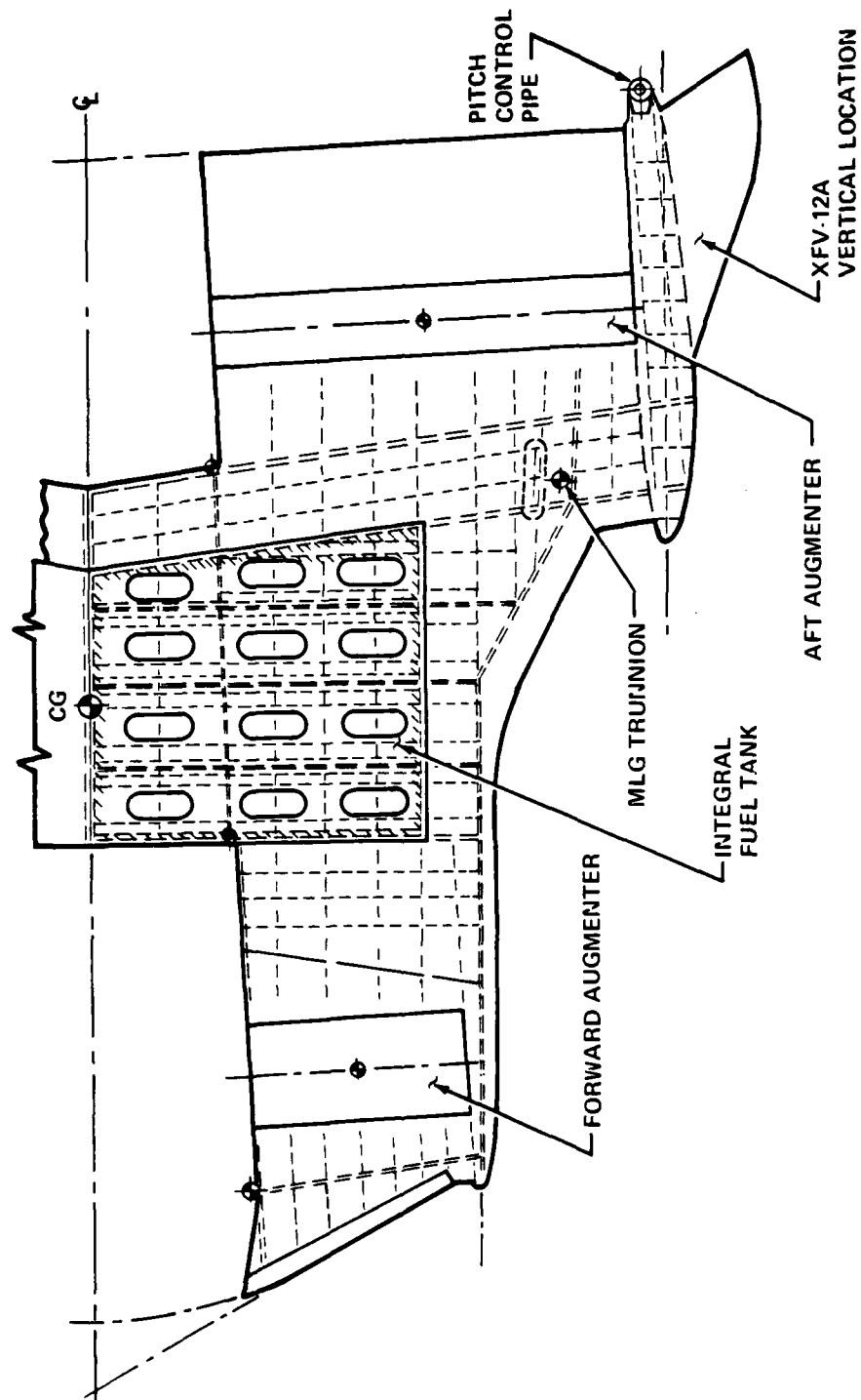
Figure 4-29 Wing Structure

COL 20926A

FORM 351-F   REV 5-77
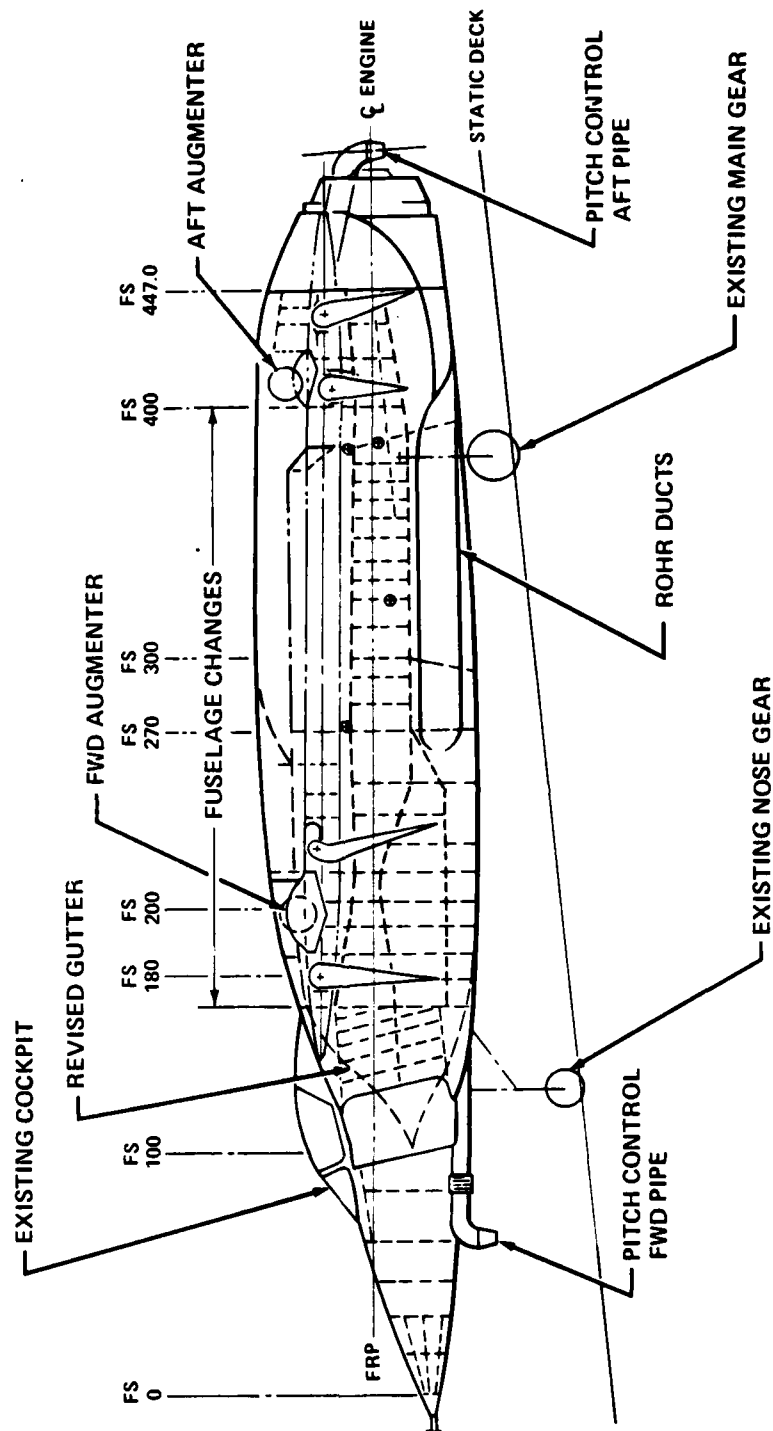
Figure 4-30 Fuselage Structure

COL 20931A

4-30

used to evaluate vertical hover time. Figure 4-31 summarizes the XFV-12B test aircraft lift using the duct loss techniques derived from the XFV-12A prototype Langley tests. A lift reserve of 1221 pounds is allocated for simultaneous control and vertical acceleration of 1.05 a/g. The resulting vertical takeoff gross weight of 20,476 pounds provides more than fifteen minutes of vertical hover time at intermediate engine power on a standard day, which is twice the test time originally predicted for the XFV-12A prototype.

4.4.3   Rectangular Augmenter Study

A review of the estimated cost through full scale tests together with the Navy's concern for the level of transition performance of the blended wing configuration, precipitated a design study of incorporating rectangular augmenters in a configuration which maintained the current wing and canard surface locations. This configuration, designated the XFV-12C, was developed using the criteria of Table 4-2 and the design philosophy of the XFV-12B. To balance the canard area increase, an increase in the wing span was required to maintain longitudinal stability. The main gears were relocated inboard on the wing box to retain the wing-fuselage attach structure, reducing the wing anhedral angle to 8.8 degrees. A comparison of the XFV-12C configuration with the XFV-12A prototype configuration is presented in Figure 4-32. Packaging of the rectangular augmenters was approached with the same philosphy as described earlier. Figures 4-33 and 4-34 show the inboard and outboard sections of the resultant wing and canard augmenters. A symmetric centerbody nozzle was incorporated into this study for comparison and to obtain the highest vertical lift possible. A comparison of the augmenter parameters is shown in Table 4-4.

The ducting system shown in Figures 4-35 and 4-36 is very similar to the XFV-12B system except for the canard ducting. Installation of the canard ducting required the canard augmenter to be located outboard of the XFV-12A position. The structural arrangement of the wing and fuselage are shown in Figures 4-37 and 4-38. Modifications to the fuselage are less extensive than with the XFV-12B study but more complex in the canard attachment. Using the weight estimating procedure described earlier, an operational weight empty of 16,963 pounds was derived. The weight and 400 pounds of fuel reserve were used to evaluate vertical hover time of the XFV-12C. Figure 4-39 itemizes the XFV-12C test aircraft lift using the duct loss techniques derived from the XFV-12A test program. Because of the geometric relationships of the canard and wing augmenters, less wing lift is required to balance the same canard lift of the XFV-12B study; therefore, less total aircraft lift is developed. This geometric relationship also requires a higher lift allowance for simultaneous control and vertical acceleration for approximately the same weight, A comparison of the vertical hover time, with and without a reaction control system, is shown to confirm that the additional 1000 pounds of useable lift more than offsets the 300 pounds of additional weight (included in the OWE of 16,963) and system complexity.
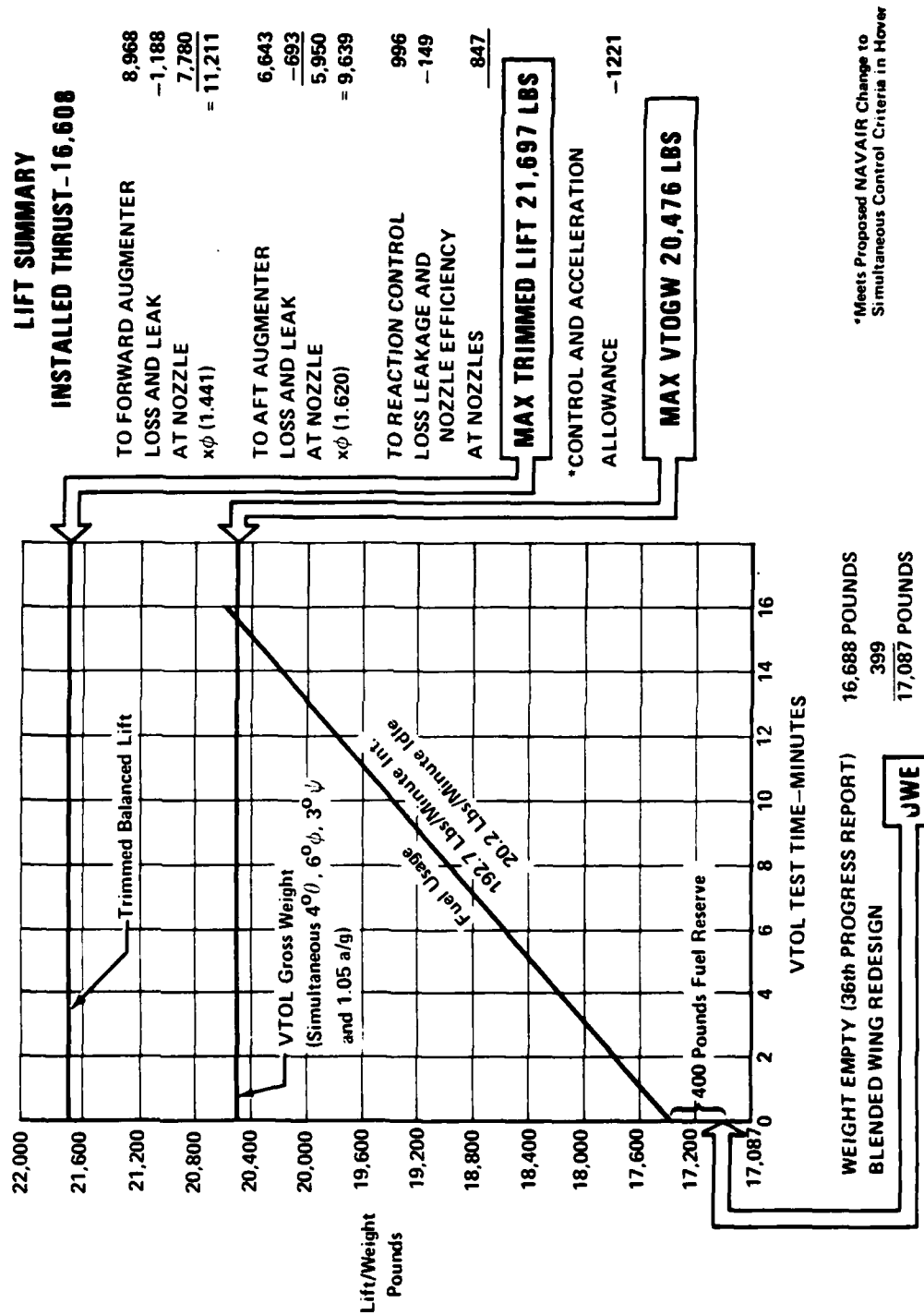
Figure 4-31 Summary of Expected Test Mission Performance

LIFT SUMMARY

INSTALLED THRUST–16,608

TO FORWARD AUGMENTER	8,968
LOSS AND LEAK	-1,188
AT NOZZLE	7,780
$x\phi$ (1.441)	= 11,211
TO AFT AUGMENTER	6,643
LOSS AND LEAK	-693
AT NOZZLE	5,950
$x\phi$ (1.620)	= 9,639
TO REACTION CONTROL	996
LOSS LEAKAGE AND	-149
NOZZLE EFFICIENCY	
AT NOZZLES	847

MAX TRIMMED LIFT 21,697 LBS

*CONTROL AND ACCELERATION
ALLOWANCE   -1221

MAX VTOGW 20,476 LBS

*Meets Proposed NAVAIR Change to
Simultaneous Control Criteria in Hover

Trimmed Balanced Lift

VTOL Gross Weight
(Simultaneous 4°, 6° ϕ, 3° ψ
and 1.05 a/g)

Fuel Usage
19.7 Lbs/Minute Int.
20.2 Lbs/Minute Idle

400 Pounds Fuel Reserve

17,087

Lift/Weight
Pounds

VTOL TEST TIME–MINUTES

WEIGHT EMPTY (36th PROGRESS REPORT)   16,688 POUNDS
                                         399
BLENDED WING REDESIGN   JWE   17,087 POUNDS

COL 20944A

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Figure 4-32   Rectangular Augmenters

CB Hinge
Y_W 140.629
Z_W 10.80

Y_W 150.209
Z_W 0.950

Y_W 155.750
Z_W 0.950

Y_W 140.629
Z_W 11.00

16°

23.40
Const

INBOARD SECTION

16°

16°

16°

OUTBOARD SECTION

Y_W 118.10
Z_W -5.920

Y_W 123.641
Z_W -5.920
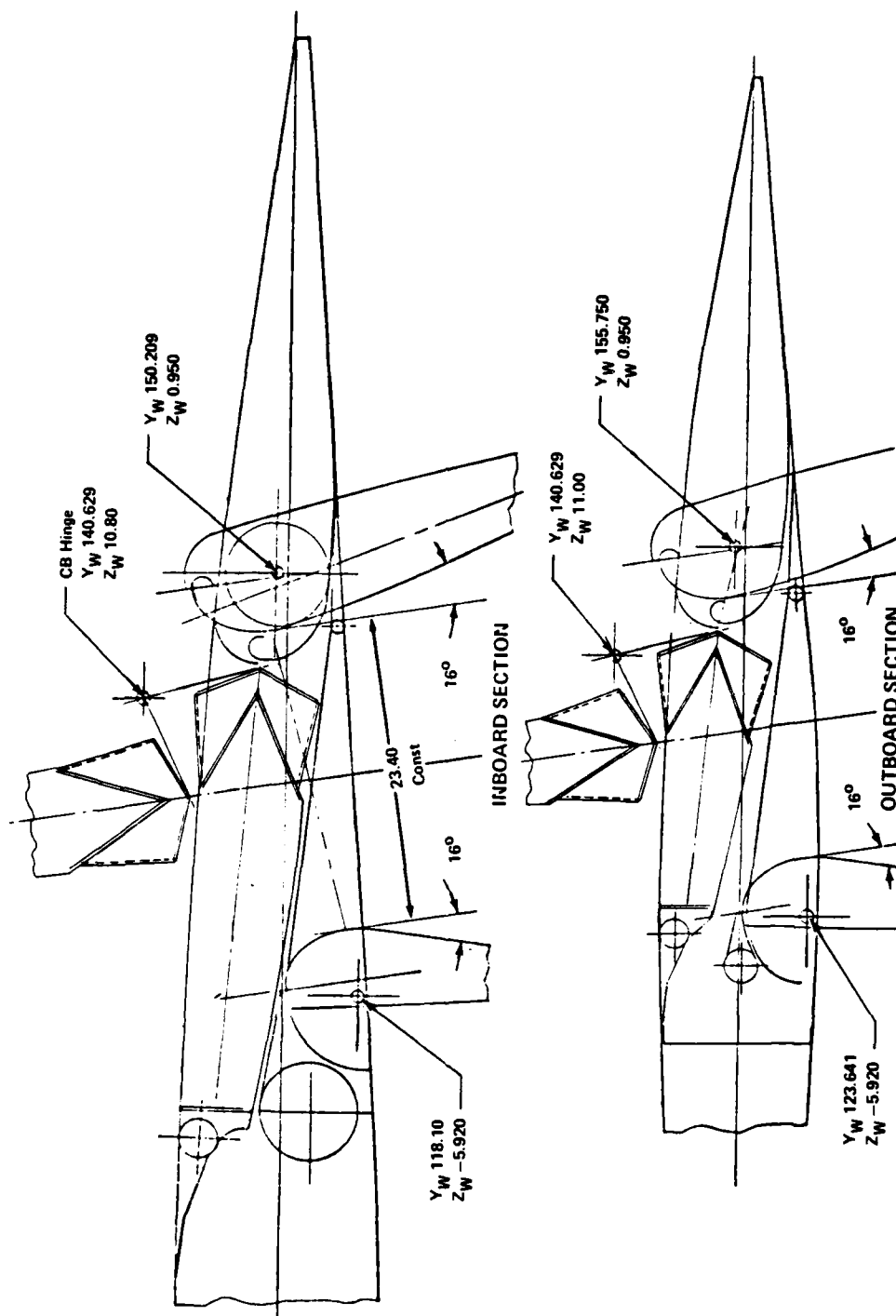
COL 22300

Figure 4-33. Wing Augmenter Cross-Sections

4-34

CB Hinge
Y_F 206.53
Z_F −22.190

Y_F 219.90
Z_F −34.80

16°

32.887
Const

16°

INBOARD SECTION

Front Spar

Y_F 176.47
Z_F −39.25

Y_F 206.53
Z_F −22.25

Y_F 219.90
Z_F −34.550

16°

OUTBOARD SECTION

16°

Front Spar

Y_F 176.47
Z_F −39.00
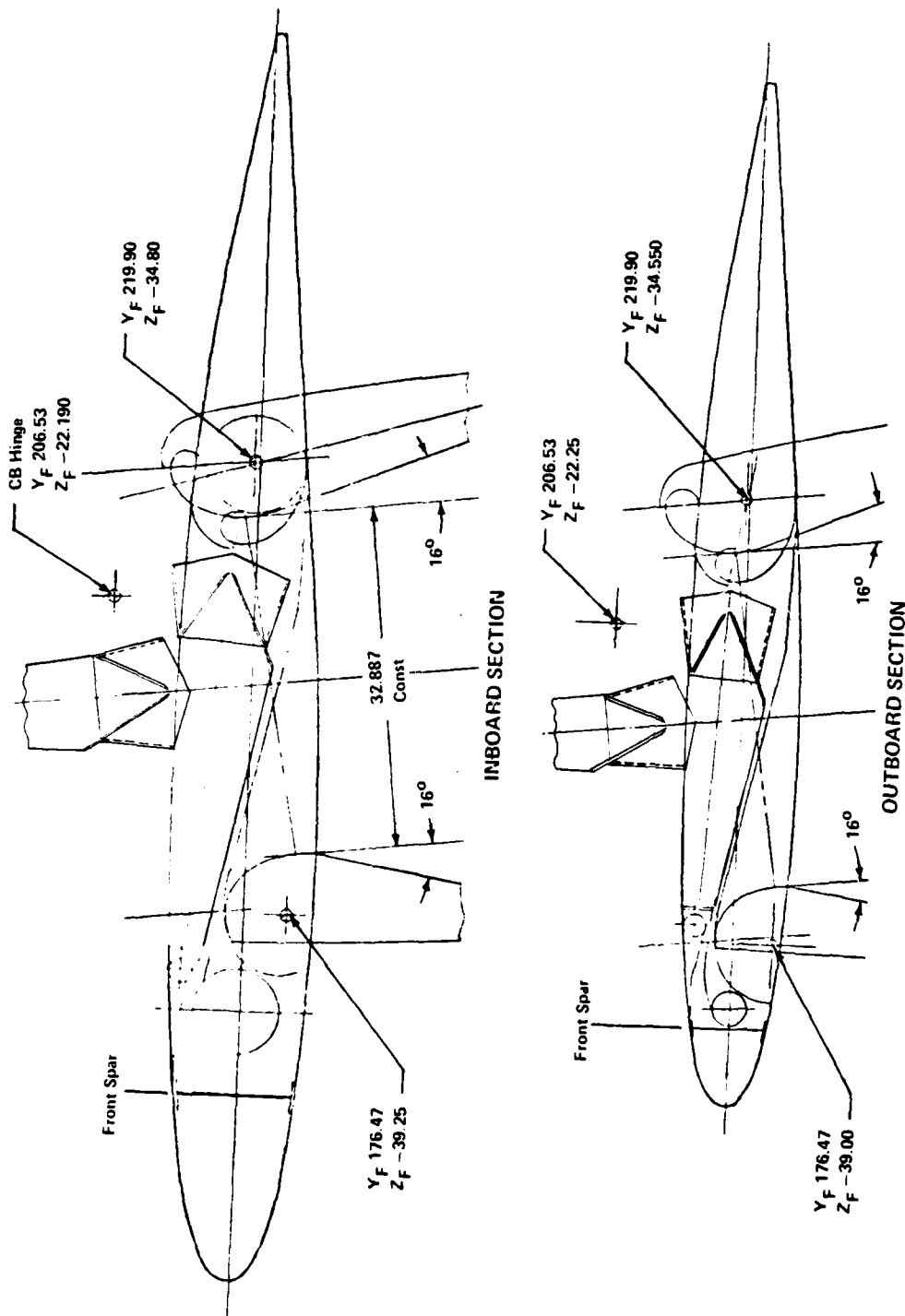
COL 22301

Figure 4-34. Canard Augmenter Cross-Sections

Forward Augmenter

PARAMETER	XFV-12C	TPA 175
A_2/A_o	13.800	13.600
A_o TOTAL	153.70 IN.^{2}	153.38 IN.^{2}
A_o DISTRIBUTION	18-60-18	15.5-63-15.5
A_2	2121.21 IN.^{2}	2086 IN.^{2}
SPAN	64.50 IN.	70.00 IN.
A_2 LOCAL	32.887 IN.	29.80 IN.
L/D	1.242	1.240
Ro/t	8.00	8.00
SPIRAL-K	1.60	DIFF 1.80 ELEV 1.20
DUCT AREA:	--	--
DIFFUSER	55.332 IN.^{2}	90.00 IN.^{2}
C.B.	184.44 IN.^{2}	165.0 IN.^{2}
ELEVATOR	55.332 IN.^{2}	63.62 IN.^{2}

Aft Augmenter

PARAMETER	XFV-12C	CONTRACT
A_2/A_o	20.50	22.304
Ao TOTAL	136.30 IN.^{2}	121.37 IN.^{2}
Ao DISTRIBUTION	18-60-18	16.6-58.3-19.4
A_2	2794 IN.^{2}	2707 IN.^{2}
SPAN	118.428	108.80 IN.
A_2 LOCAL	23.59 IN.	25.20 IN.
L/D	1.69	1.25
Ro/t	12.00	DIFF 10.39 ELEV 12.075
SPIRAL-K	1.30	1.30
DUCT AREA:	--	--
• DIFFUSER	49.07 IN.^{2}	50.27 IN.^{2}
• C.B.	163.56 IN.^{2}	140.6 IN.^{2}
• ELEVON	49.07 IN.^{2}	50.64 IN.^{2}

Table 4-4. XFV-12C Augmenter Parameters

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Wing Augmenter Feed System

FIXED HINGED BELLOWS

FROM DIVERTER

FORWARD DIFF FEED

ELEVON FEED

UP POSITION

C.E. FEED DOWN POSITION

Canard Augmenter Feed System

THERMAL EXPANSION BELLOWS

MANIFOLD

FROM ROHR DUCT

ELEVATOR BEARINGS

ELEVATOR

CENTER EJECTOR

C.E. BEARING

FWD PITCH PIPE

SOV

FWD DIFFUSER
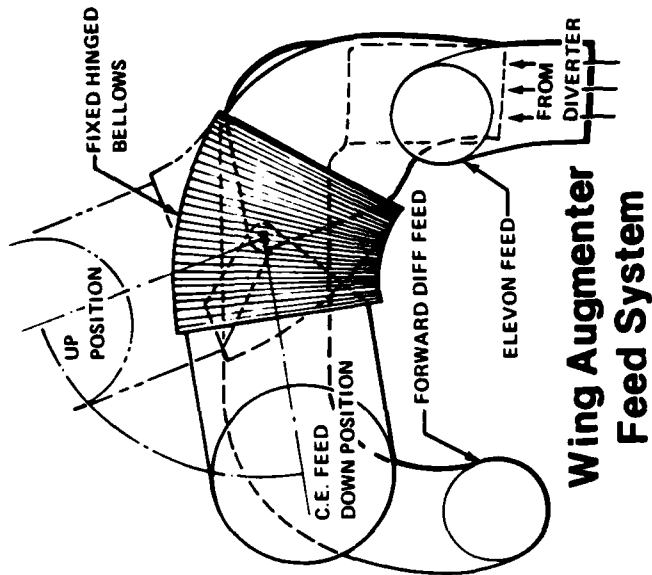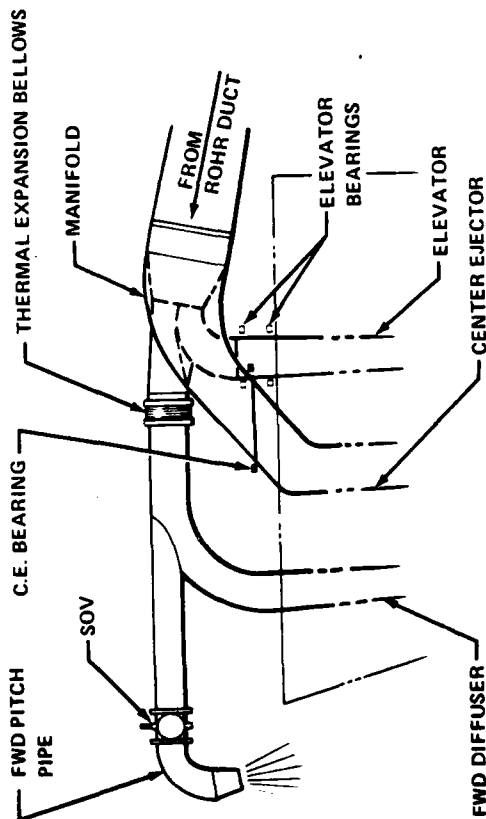
Figure 4-35   Ducting System

COL 21523
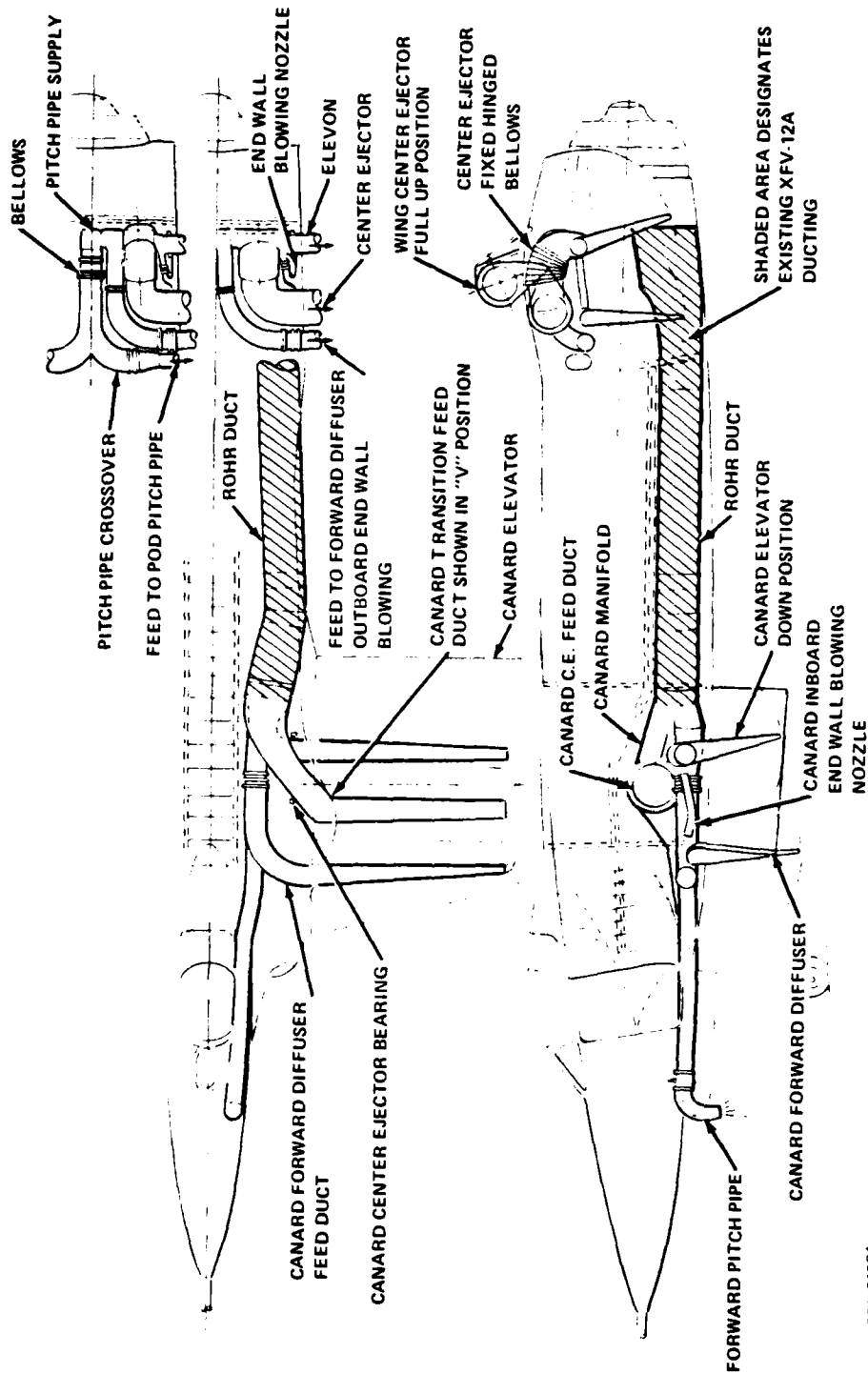
Figure 4-36. Ducting System

Labels:
BELLOWS
PITCH PIPE SUPPLY
END WALL BLOWING NOZZLE
ELEVON
CENTER EJECTOR
WING CENTER EJECTOR FULL UP POSITION
CENTER EJECTOR FIXED HINGED BELLOWS
SHADED AREA DESIGNATES EXISTING XFV-12A DUCTING
PITCH PIPE CROSSOVER
FEED TO POD PITCH PIPE
ROHR DUCT
FEED TO FORWARD DIFFUSER OUTBOARD END WALL BLOWING
CANARD TRANSITION FEED DUCT SHOWN IN "V" POSITION
CANARD ELEVATOR
CANARD C.E. FEED DUCT
CANARD MANIFOLD
ROHR DUCT
CANARD ELEVATOR DOWN POSITION
CANARD FORWARD DIFFUSER FEED DUCT
CANARD CENTER EJECTOR BEARING
CANARD INBOARD END WALL BLOWING NOZZLE
FORWARD PITCH PIPE
CANARD FORWARD DIFFUSER

COL 21524

4-38

Figure 4-37 Wing Structure

- ¢ AIRCRAFT
- WING AUGMENTER
- PITCH CONTROL PIPE
- EXISTING XFV-12A VERTICALS
- NEW POD
- MLG TRUNNION
- INTEGRAL FUEL TANK
- FS 287.50
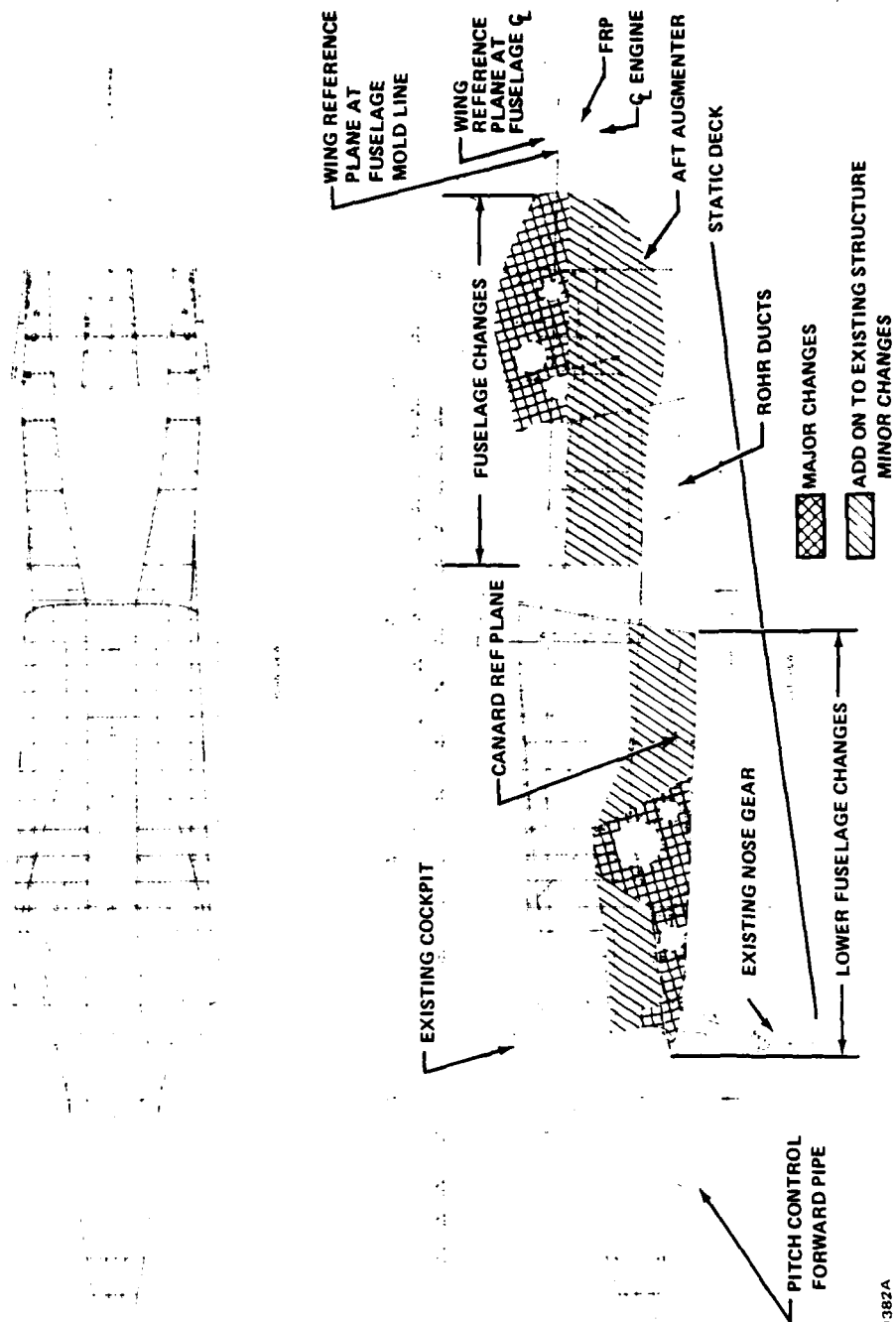
COL 21375

Figure 4-38 Fuselage Structure

COL 21382A

LIFT SUMMARY

INSTALLED THRUST - 16,608

TO CANARD AUGMENTER	8,802
LOSS AND LEAK	-1,317
AT NOZZLE	7,485
$\times \phi$ (1.45)	10,853
TO WING AUGMENTER	6,809
LOSS AND LEAK	-707
AT NOZZLE	6,102
$\times \phi$ (1.60)	9,763
TO REACTION CONTROL	996
LOSS LEAKAGE AND	-110
NOZZLE EFFICIENCY	
AT NOZZLES	886

MAX TRIMMED LIFT 21,502

*CONTROL AND ACCELERATION
ALLOWANCE   -1,302

MAX VTOGW   20,200

*Meets Proposed NAVAIR Change to
Simultaneous Control Criteria in Hover

REACTION
CONTROL

TRIMMED BALANCED LIFT

VTOL GROSS WEIGHT
(SIMULTANEOUS $4^0 \theta$, $6^0 \theta$, $\phi = 3^0 \psi$
AND 1.05 a/g)

NO REACTION CONTROL

192.7 POUNDS/MINUTE

400 Pounds Fuel Reserve

OWE

Lift/Weight
Pounds

21,200
20,800
20,400
20,000
19,600
19,200
18,800
18,400
18,000
17,600
17,200
16,800

0   2   4   6   8   10   12   14   16

VTOL TEST TIME—MINUTES

WEIGHT EMPTY (36th PROGRESS REPORT)	16,688 POUNDS
RECTANGULAR AUGMENTER REDESIGN	275
	16,963 POUNDS

COL 21512B

Figure 4-39 Summary of Expected Test Mission Performance - XFV-12C

4.5   SUMMARY

Detailed studies have shown that high performance thrust augmenter ejectors can be incorporated into a supersonic aircraft configuration with mission performance comparable with existing fleet aircraft. An empirical, analytical, and design data base has been generated that can evaluate the sensitivities of configurations and provide explicit guidelines to the designers.

High performance Coanda shapes can be incorporated into thin airfoil shapes necessary for supersonic flight. Design criteria have been established which provides the designer the necessary information to establish coanda/flap shapes that will support augmentation values ($\emptyset$) in excess of 1.60.

Centerbody nozzle configurations can be integrated into the wing augmenter that will provide augmentation values over a range of 1.46 to 1.64.

The capability to package rectangular augmenters in a viable demonstrator configuration exists as shown by either study – the XFV-12B or the XFV-12C. Technical concerns exist for both configurations; however, the XFV-12B configuration is more representative of an operational configuration. The level of performance during transition from vertical flight to conventional flight is the primary concern for this configuration and should be demonstrated prior to development. The XFV-12B configuration has the highest vertical lift capability along with excellent ground effects, reingestion, and control characteristics and provides the pilot the greatest peripheral vision during vertical flight. The XFV-12C configuration's transition performance capability has been demonstrated by comparison; however, the level of vertical lift capability in ground effect is unknown and should be demonstrated prior to development.

5.0 MODELING

5.1 GENERAL

In order to compensate for the deficiencies of any experiment and the limitations of any analysis, the ejector development effort was carried out using both these tools. Numerical simulation was used to study the development of the jets from proposed centerbody nozzles and to predict the ideal pressure distributions on the ejector shroud. Scale model tests were used to verify predicted performance trends and to provide data for checking and improving the analytical methods. However, since the ejector and the analysis were being developed simultaneously, discrepancies between calculation and measurement were studied critically, in order to identify practical limits on ejector performance, as well as theoretical deficiencies.

Both the numerical and experimental models employed in the XFV-12A development program are presented here. In this section, the analytical methods used for predicting the entrainment of the turbulent jets and the pressure distribution on the ejector flaps are described. The ejector scaling parameters and the experimental apparatus and test procedures are also presented.

5.2 NUMERICAL MODELS

In order to compute the thrust augmentation without solving the full Navier-Stokes equations, some approximations must be made. The classical approach utilizes streamwise integration of the governing equations to determine the augmentation as a function of the jet mixing rate. The solution is obtained by iterating on the inlet velocity until the exhaust pressure matches the ambient pressure outside the ejector. This approach formed the basis of the method used for analyzing the development of the jets from various nozzles. However, it was necessary to devise an approach based on airfoil theory in order to calculate the surface pressure distribution on the airfoil flap. Both methods of analysis will be described in this section.

5.2.1
JET MIXING ANALYSIS

In order to predict the jet flow fields which develop from the complex nozzles being studied, it is necessary to determine the solution for a turbulent, three-dimensional velocity and pressure field. Considerable savings in computer storage and running time were achieved by utilizing a procedure developed by Patankar and Spalding (1972), Reference (7) to reduce solution of the original boundary value problem to an initial value problem which may be solved by streamwise marching procedures. Since there is a primary direction of flow (through the ejector), the streamwise velocity component is considered to be driven by a mean pressure $\bar{p}(x)$, which is decoupled from the perturbation pressures, $p'(x,y,z)$ in the

transverse planes. In addition, the gradients of the normal stresses are neglected. Under these assumptions, the governing equations in Cartesian coordinates become

Equation of State

$$\rho = \frac{\overline{P}}{RT}$$

Continuity

$$\frac{\partial}{\partial X}(\rho U) + \frac{\partial}{\partial Y}(\rho V) + \frac{\partial}{\partial Z}(\rho W) = 0$$

Momentum

$$\frac{\partial}{\partial X}(\rho U^2) + \frac{\partial}{\partial Y}(\rho UV) + \frac{\partial}{\partial Z}(\rho UW) = \frac{\partial \tau_{YX}}{\partial Y} + \frac{\partial \tau_{ZX}}{\partial Z} - \frac{\partial \overline{P}}{\partial X}$$

$$\frac{\partial}{\partial X}(\rho UV) + \frac{\partial}{\partial Y}(\rho V^2) + \frac{\partial}{\partial Z}(\rho VW) = \frac{\partial \tau_{YY}}{\partial Y} + \frac{\partial \tau_{ZY}}{\partial Z} - \frac{\partial P'}{\partial Y}$$

$$\frac{\partial}{\partial X}(\rho UW) + \frac{\partial}{\partial Y}(\rho VW) + \frac{\partial}{\partial Z}(\rho W^2) = \frac{\partial \tau_{YZ}}{\partial Y} + \frac{\partial \tau_{ZZ}}{\partial Z} - \frac{\partial P'}{\partial Z}$$

Energy

$$\frac{\partial}{\partial X}(\rho UH) + \frac{\partial}{\partial Y}(\rho VH) + \frac{\partial}{\partial Z}(\rho WH) = \frac{\partial}{\partial Y}\left(\frac{\mu_t}{Pr}\frac{\partial H}{\partial Y}\right) + \frac{\partial}{\partial Z}\left(\frac{\mu_t}{Pr}\frac{\partial H}{\partial Z}\right)$$

Here, U, V, W, are the time averaged velocity components and the τ_{ij} are the turbulent shear stresses.

The turbulent stresses were related to the mean velocity gradient according to the usual eddy viscosity assumption (Schlicting 1968), Reference (8). The eddy viscosity was calculated using the turbulence kinetic energy model of Launder and Spalding (1974), Reference (9). According to this model, the eddy viscosity is assumed to be a function of the turbulence kinetic energy, k, and its rate of dissipation, ε . Dimensional analysis gives

$$\mu_t = C_\mu \frac{\rho k^2}{\varepsilon}$$

in which C_μ is a constant. The transport equations for k and ε are

$$\frac{\partial(\rho Uk)}{\partial X} + \frac{\partial(\rho Vk)}{\partial Y} + \frac{\partial(\rho Wk)}{\partial Z} = \frac{\partial}{\partial Y}\left(\frac{\mu_t}{\sigma_k}\frac{\partial k}{\partial Y}\right) + \frac{\partial}{\partial Z}\left(\frac{\mu_t}{\sigma_k}\frac{\partial k}{\partial Z}\right) + \mu_t\left[\left(\frac{\partial U}{\partial Y}\right)^2 + \left(\frac{\partial U}{\partial Z}\right)^2\right] - \rho\varepsilon$$

$$\frac{\partial(\rho U\varepsilon)}{\partial X} + \frac{\partial(\rho V\varepsilon)}{\partial Y} + \frac{\partial(\rho W\varepsilon)}{\partial Z} = \frac{\partial}{\partial Y}\left(\frac{\mu_t}{\sigma_\varepsilon}\frac{\partial \varepsilon}{\partial Y}\right) + \frac{\partial}{\partial Z}\left(\frac{\mu_t}{\sigma_\varepsilon}\frac{\partial \varepsilon}{\partial Z}\right) + c_1\mu_t\left[\left(\frac{\partial U}{\partial Y}\right)^2 + \left(\frac{\partial U}{\partial Z}\right)^2\right] - c_2\rho\varepsilon\left]\frac{\varepsilon}{k}\right)$$

The values of the constants, C_μ, σ_k, σ_ϵ, C_1, C_2, suggested by Launder and Spalding were also used here.

These equations were put in finite difference form by integrating them over a control volume surrounding a typical grid point. A more complete description of this procedure and an illustration of its use has been given by DeJoode and Patankar (1978), Reference (10).

Since these equations are solved by streamwise integration, initial values of all the flow variables must be specified in order to start the calculation. The initial jet velocity was calculated from the isentropic velocity at the specified pressure ratio. It was assumed that the internal, viscous losses were the same for each nozzle, so that the same velocity coefficient, $C_V = 0.925$ was used in each case. Because most of the nozzles to be studied employ some jet deflection to generate hypermixing vortices, the initial, calculated jet velocity vector is inclined to the ejector axis. Thus, the predictions of ejector thrust augmentation due to increased mixing are balanced against the tilt loss in the primary jet thrust. The angularity of the Coanda jets and secondary entrained flow is similarly treated.

The initial value of the turbulence kinetic energy is expressed as a fraction of the mean flow energy,

$$k = C_k U_j^2$$

and the initial dissipation rate is assumed to be proportional to the rate of energy addition by the jet

$$\epsilon = C_\epsilon U_j^3/t$$

in which U_j and t are the initial values of the jet velocity and width. The values of C_k and C_ϵ were determined from available data. As reported by Tennekes and Lumley (1972), Reference (11), measurements in many jet and wake flows indicate that the turbulence energy is correlated with the turbulent stress,

$$k = \overline{u'v'}/0.4$$

In self preserving flows, the turbulent stress is proportional to the mean velocity. The maximum stress is given by

$$\overline{u'v'} = U_j^2/R_T \ e^{\frac{1}{2}}$$

in which R_T is an empirical constant which depends on the type of flow; for plane jets, $R_T = 26$. Combining these relations yields $C_k = 0.06$. Similarly, the dissipations in self preserving flows is given by

$$\epsilon = U_j^3/t \ R_T \ e^{\frac{1}{2}}$$

which yields $C_\epsilon = 0.02$. Additional details regarding the initial conditions subroutine have been given by Mefferd and Bevilaqua (1978) Reference (12).

The computational time required to obtain a solution for a typical ejector configuration is on the order of 15 minutes. However, it originally took more than 20 man-hours to set up the finite difference grid and specify all the values of the velocity and turbulence components at each grid point. In order to reduce this time and eliminate the arbitrariness in the grid specification, a computer code was developed as part of the XFV-12A program to generate this information from the basic ejector geometry and specified grid spacing rules. The interactive capabilities of the CYBER 176 computer and the Tektronic 4014 graphics terminal can be utilized to modify the grid, if desired. The grid generated for a typical multi-lobe nozzle is shown in Figure 5-1. The high grid density required in the wall jets is apparent. Because the nozzle is periodic along the ejector span, symmetry planes are used as computational boundaries. Details of this code and a step-by-step procedure for its use are given by Long (1981), Reference (13). use of this code has reduced the set-up time to less than an hour.

The thrust augmentation ratio, ϕ, is evaluated from the calculated momentum flux at the ejector exit. The iteration procedure is converged to within $\Delta\phi = \pm.02$. Greater accuracy could have been achieved at some cost in computer running time, but differences in augmentation of this magnitude were comparable to the experimental error

5.2.2   EJECTOR POTENTIAL FLOW ANALYSIS

The increase in thrust that results from the turbulent mixing of the primary jet and secondary streams appears as a reaction force on the ejector shroud. The reaction force is the integral of the surface pressure distribution on the shroud. In order to evaluate the measured pressure distribution it is necessary to compute the ideal pressure distributions. A method for calculating these distributions will be described in this section

The shroud can be described as "flying" in the velocity field of the flow entrained by the jet, thus, it experiences a force analogous to the lift developed on a wing fixed in a moving stream. The circulation generated around each section of the ejector shroud can be calculated by solving a system of equations which specify that the shroud must be a streamline of the flow induced by the entrainment of the jets. For this calculation, the strengths of the sinks which represent each jet are considered to be known. These strengths can be determined from the turbulent mixing of the jets computed in the viscous solution previously discussed, or from experimental data.

A potential flow solution for the ejector wing has been obtained by replacing the shroud elements and jets with equivalent flow singularities. The flow is assumed to be irrotational, incompressible and attached. Since the flow satisfies Laplace's equation, the flow
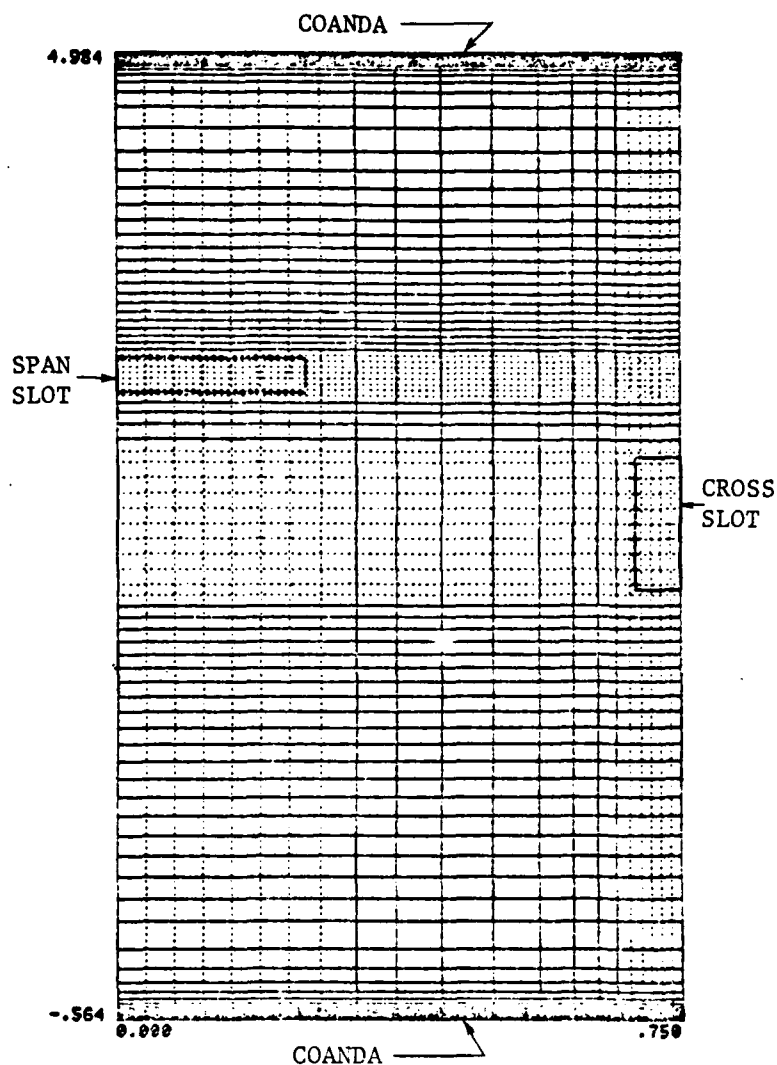
Figure 5-1   Computational Grid for Cross-Slot –
Span Slot Centerbody Configuration

velocities at any field point may be obtained by superimposing the induced
velocity due to all the flow singularities. The tangential and normal
components of the resultant velocity at the panel control points may be
expressed as follows:

$$V_{t_j} = \Sigma\, A_{s_{ji}}\, \sigma_i + \Sigma\, A_{v_{ji}}\, \gamma_i$$

$$V_{n_j} = \Sigma\, B_{s_{ji}}\, \sigma_i + \Sigma\, B_{v_{ji}}\, \gamma_j$$

where σ and γ are the source and vortex strengths respectively and A_s,
A_v, B_s, B_v are the influence coefficients which depend only upon
panel geometry.

Rockwell International

The shroud surfaces are shown in Figure 5-2 to be represented by a series
of curved parabolic panels. The continuous distribution of flow
singularity densities corresponding to an exact distribution is
approximated by piecewise linear distributions of both source and vortex
densities on each curved panel. The sink strengths and distributions are
used to simulate jet entrainment. Vortex densities are the only unknowns
in this potential flow formulation and these can be determined by
appropriate boundary conditions on the ejector wing elements.

Application of the Newman boundary condition, that is, specifying the
normal velocity at each control point of the panel elements, results in a
system of linear equations in the unknown vortex strengths, γ. Having
determined the singularity strengths by solving these equations, the panel
velocities, pressures and off body velocities can then be obtained.
Details of this inviscid solution for an ejector wing were presented by
Bevilaqua and DeJoode (1978), Reference (1), for the static case, and by
Bevilaqua, Woan, and Schum (1981), Reference (14), for the case of an
ejector in forward flight.

5.3   EJECTOR SCALING LAWS

Although numerical simulation facilitates the development of some ejector
components, the complexity of the ejector flow field requires that
complementary testing be performed during the development process. There
are some advantages to performing these tests with full size components
and hot gas jets but, for reasons of cost and convenience, small scale
models driven by cold air jets were used. While such models can only
approximate the performance of full size ejectors, the approximation can
be quite close if the appropriate scaling laws are satisfied. The purpose
of this section is to discuss the requirements for ejector scaling and to
show that analysis and scale model testing can be used to design full size
ejectors.

In order to set up a meaningful test, the fundamental parameters on which
the thrust augmenting force depends must be identified. If it is assumed
that this force is a function of the jet thrust, the ejector duct
geometry, and the physical properties of the fluid, dimensional analysis
yields for the scaling law

$$F/T = f (Re, M, L/W, \delta_F)$$

in which the force coefficient, F/T, is the ratio of the augmenting force
to the jet thrust, Re and M are the jet Reynolds number and Mach number,
and L/W and δ give the length to width ratio and diffuser half angle of
the ejector duct. The force coefficient also depends on the surface
roughness and ambient turbulence level, but these parameters should be
controlled to insure that their effect is small. Temperature effects are
usually assumed to be implicit in the variation of the Mach and Reynolds
numbers; however, there is an additional effect of temperature on the
turbulent mixing which will be discussed separately.
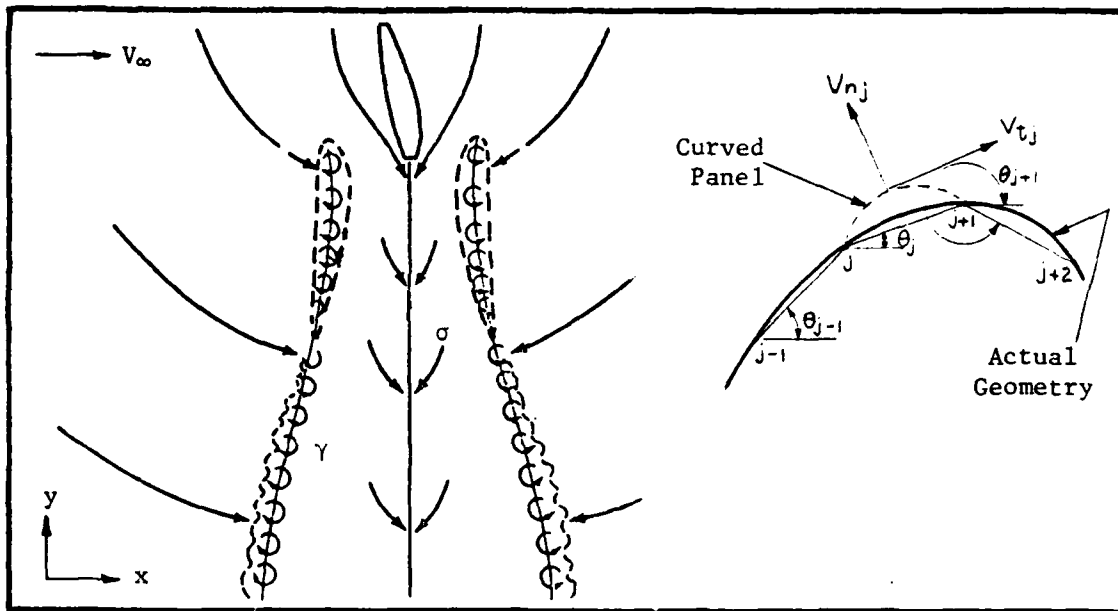
Figure 5-2 Mathematical Model of the Ejector Wing

The geometry of a scale model can be made to duplicate the full size prototype, but it is not possible to simultaneously match the Mach and Reynolds numbers of the prototype. For example, if the Mach numbers are matched, then the change in scale means the model Reynolds number will be smaller. Similarly, if the Reynolds numbers are matched by increasing the model velocity, then the model Mach number will be larger. However, because the velocity of the prototype jet is large (M > 1), the Reynolds number is also large (Re $\approx 10^6$); consequently, the flow is turbulent and the effects of viscosity are small. In this case, changes in the Reynolds number only affect the very smallest scales of the turbulence, which do not interact directly with the main flow. According to this principle of asymptotic invariance, the Reynolds number is not a relevant parameter if its value is large.

Therefore, if the Mach numbers are matched and the Reynolds numbers are large, scale model tests can be used to determine the variation of ejector thrust with nozzle geometry and diffuser angle for a given ejector configuration. On the other hand, the angle at which the flow separates, and other phenomena related to the exact details of the viscous stresses, are dependent on the Reynolds number. Thus, unless the Reynolds number is matched, model values of the separation angle cannot always be used to predict full scale separation angles. However, separation will occur at a smaller angle on the scale model, so that assuming that the prototype separates at the same angle as the model will be conservative.

As previously noted, the first order effects of temperature on the
physical properties of the jet (density, viscosity, and compressibility)
are implicit in scaling the Mach and Reynolds numbers. In particular, the
variation of Reynolds number with temperature is shown in Figure 5-3. The
effect of reducing the jet temperature by 1000°F, as in the present case,
does not change the order of magnitude of the Reynolds number or the
turbulent character of the flow. On the other hand, the Mach number

$$P_{AMB} = 14.7 \ PSIA$$

NPR
2.2:1
2.0:1
1.8:1
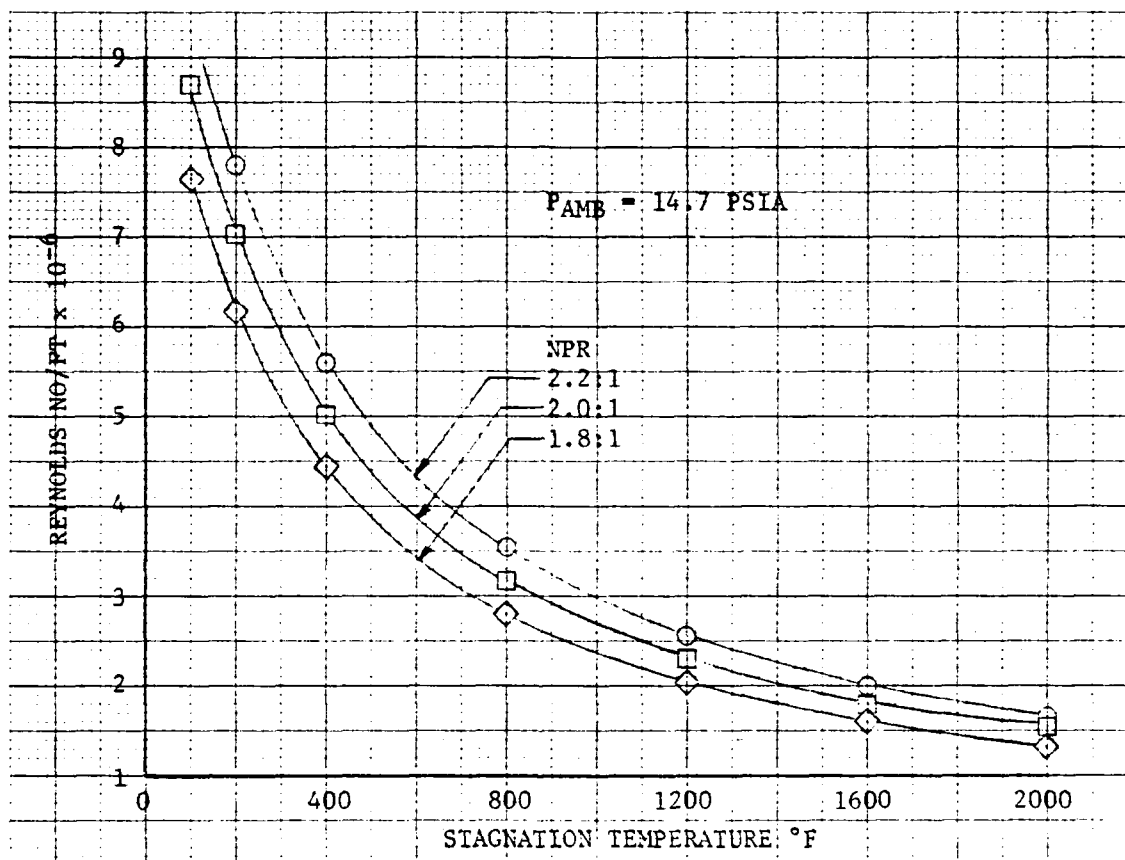
Figure 5-3   Reynolds No. Variation With Temperature

($M \equiv v/a$) is independent of temperature, since both the jet velocity and
the speed of sound, a, have the same dependence on temperature:

jet velocity $\qquad V = [\frac{2\gamma}{\gamma-1} RT_0 \ (1 - (\frac{P}{P_0})^{\frac{\gamma-1}{\gamma}})]^{\frac{1}{2}}$

speed of sound $\qquad a = (\gamma RT_0)^{\frac{1}{2}}$

The jet thrust is also independent of temperature since the velocity
increase is balanced by a decrease in density.

The effect of temperature on the rate of turbulent mixing is not as
straightforward. For small density differences, the mixing rate is

proportional to the velocity difference between the two streams. But, if the density difference is large, the mixing rate is proportional to the momentum difference. Thus, a small temperature difference will probably increase the mixing rate ($\Delta V \sim T_o^{1/2}$) while a large temperature change will probably decrease mixing ($\Delta V \sim T_o^{-1/2}$), although this has not been proven.

The net effect of the 1000°F temperature decrease used for these scale model tests was calculated with the jet mixing program to be a .03 increase in augmentation. This is almost within the accuracy of the calculation procedure; however, the available data seems to support this result. This is shown in Figure 5-4. Although all the data except that
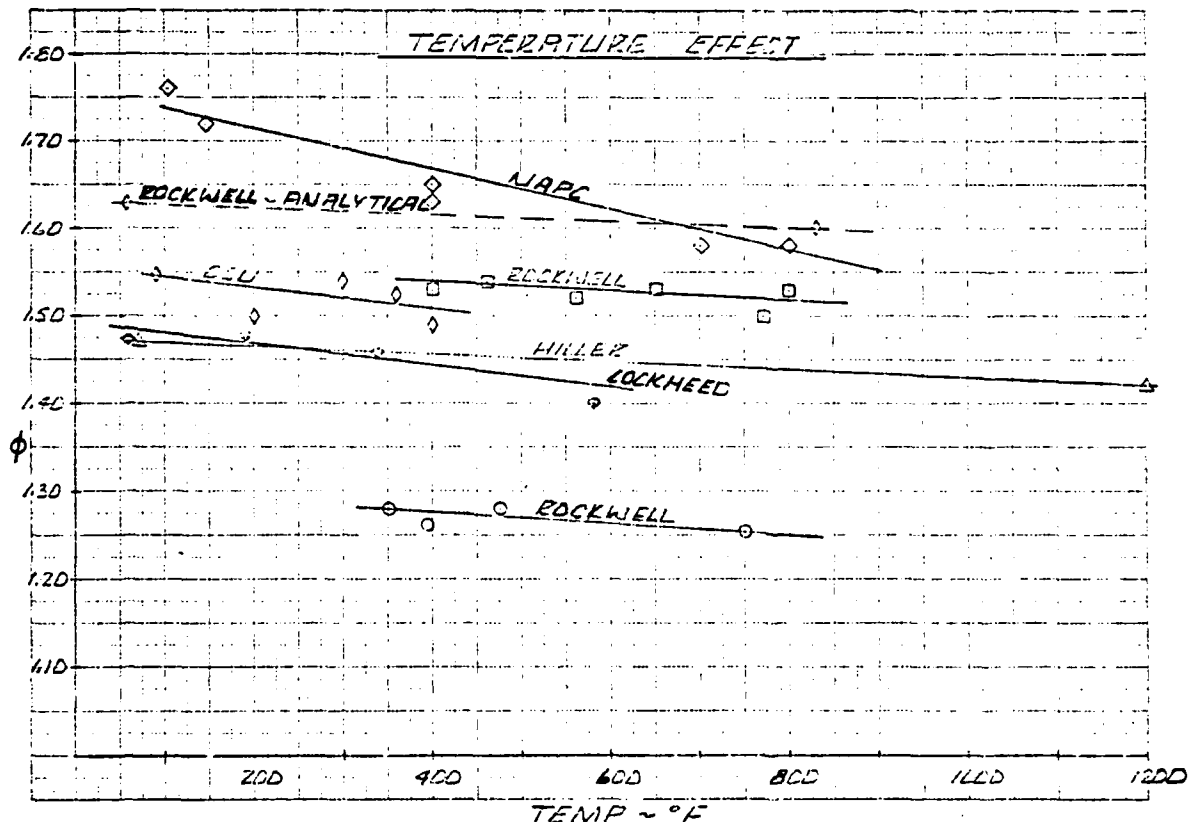
Figure 5-4 Temperature Effect

of Lockheed also included a scale or configuration change, the trend is consistent. Thus, a small increase in performance may be expected as a result of using cold jets in the model.

Full size ejectors built by Lockheed, Boeing, and DeHavilland aircraft companies produced less augmentation than the laboratory models from which they were developed. However, there were differences between the model and full size ejectors in construction, jet temperature, and various ejector parameters, so that these results are inconclusive regarding scale effects. Full size ejectors tested by Rockwell International have

generally performed as well as the laboratory models and, in one case for which the geometric scaling was nearly exact, the same performance was obtained (Mefferd, Alden and Bevilaqua, 1979), Reference (22). These ejectors are shown in Figure 5-5. Thus, it is felt that scale model testing can be used, with some caution in designing the test and interpreting the results, for the development of aircraft ejectors.

5.4   APPARATUS

All scale model augmenter and isolated centerbody nozzle testing was conducted at Rockwell International's NAAD Columbus Thermodynamic Laboratory. During this scale model test program, two test stands, designated as Rig numbers 1 and 5 were utilized. The two rigs are similar in design and are basically load test stands capable of measuring lift forces to 1000 pounds, drag loads to 200 pounds, and side forces up to 2000 pounds. It should be noted that Rig No. 1 was a 2-D stand capable of measuring only lift and drag components. However, considering the nature of the hardware under evaluation in this phase of testing, a 2-D stand was found to be adequate to satisfy test requirements.

Each test stand is basically a floating frame, attached to a fixed outer frame through load cells to measure the two or three orthogonal forces. Tension is maintained on the load cells with cable suspended preloads. Air is supplied to the augmenter models from plant air or from the Thermodynamic Laboratory's compressor through four individually controlled lines. The air supply system is equipped with particle filters and dryers. Excessive variations in supply air temperature are controlled with the use of a cooling tower.

Each supply has a venturi meter with the necessary instrumentation attached to calculate mass flows using standard venturi meter equations. The four lines are fed to the diffuser, centerbody, elevon, and endwall blowers (BLCS) through four flexible two inch hoses. These flexible lines prevent excessive tare when bringing the lines across the metric part of the load stand. Tares in the lift direction are typically one to two pounds, and repeatable, and are accounted for in the data reduction program. Appropriate valving is incorporated in the plumbing so the venturis may be used with any test rig, although not at the same time. Figures 5-6, 5-7, and 5-8 are sketches of the air supply system and test stand No. 1 details, while Figure 5-9 is an overall view photograph of test stand No. 1 with a typical model installed.

Load cells, pressure transducers and thermocouples are connected at appropriate test points on the load stand, model, and air supply system to monitor forces, pressures, and temperatures. These transducers are periodically calibrated using standard laboratory practice with calibration equipment traceable to the National Bureau of Standards.

The Thermo Lab's data system is capable of monitoring 48 channels of transducer analog output data and incorporates all signal conditioning and amplification prior to signal processing by an IBM 1800 computer. The data system also has the capability of performing an electrical test of the transudcer's bridge integrity by use of a resistance calibration test
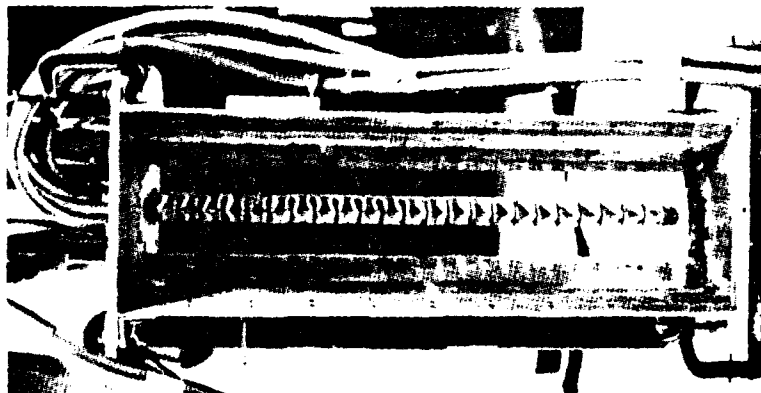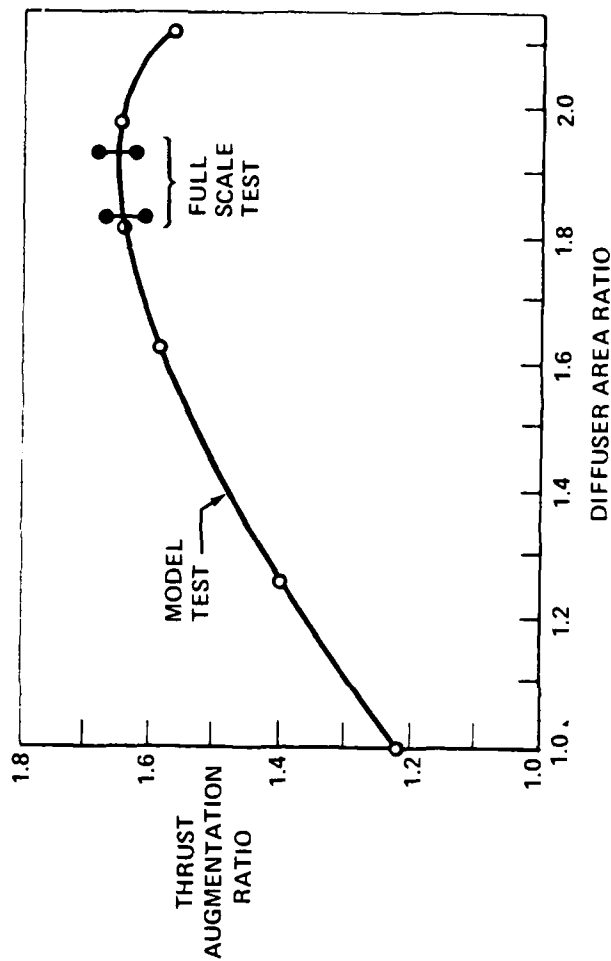
Figure 5-5. Scale and Full Size Ejectors

FROM PLANT AIR

FROM COMPRESSOR

4" MANIFOLD

2" LINE

VENTURI

UNIFLOW
VALVE

GATE VALVE

TO ALT. RIG

TO BLC

TO ALT. RIG

TO C/B

TO ALT. RIG

TO DIFF

TO ALT. RIG

TO ELEV
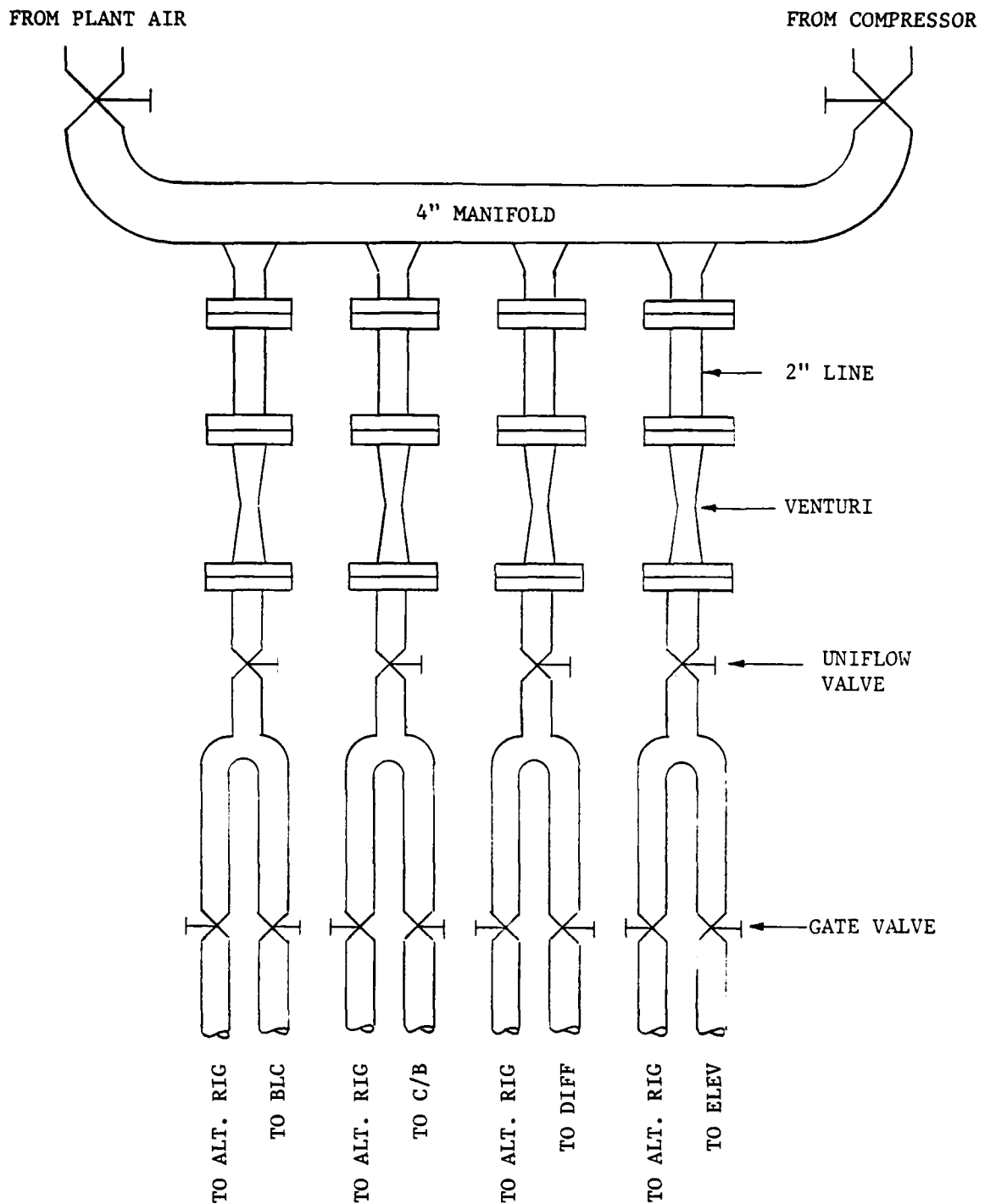
Figure 5-6. Air Supply System
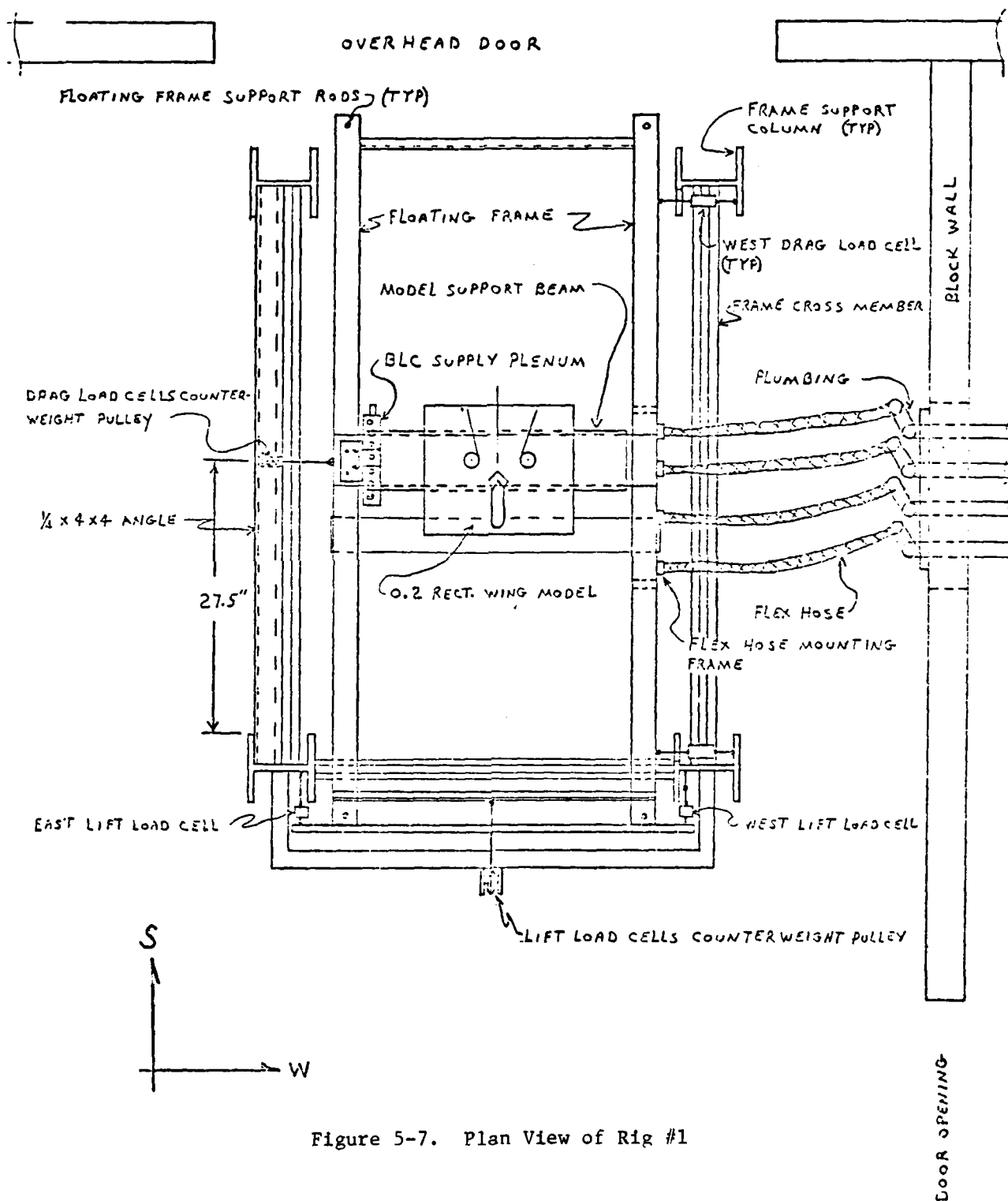
Figure 5-7.   Plan View of Rig #1

Rockwell International

½ ⌄ 4 X 4 ANGLE

PLENUM BRACKET

BLC PLENUM

CLEVIS LINKG

DRAG LOAD CELL COUNTERWEIGHT CABLE PULLEY

MODEL SUPPORT BEAM

CUT OUT FOR LOAD CELL

CABLE TO 2" FROM FLOOR

FLOATING FRAME MEMBER

DRAG LOAD CELL (TYP)

FRAME SUPPORT COLUMN

FRAME CROSS MEMBER

LOOKING SOUTH

LOOKING SOUTH

FLOATING FRAME MEMBER

FLEX HOSE FRAME

TO MODEL

2" FLEX HOSE

BLOCK WALL

2" SUPPLY PIPE

Figure 5-8. Section Blowups of Modifications to Rig #1
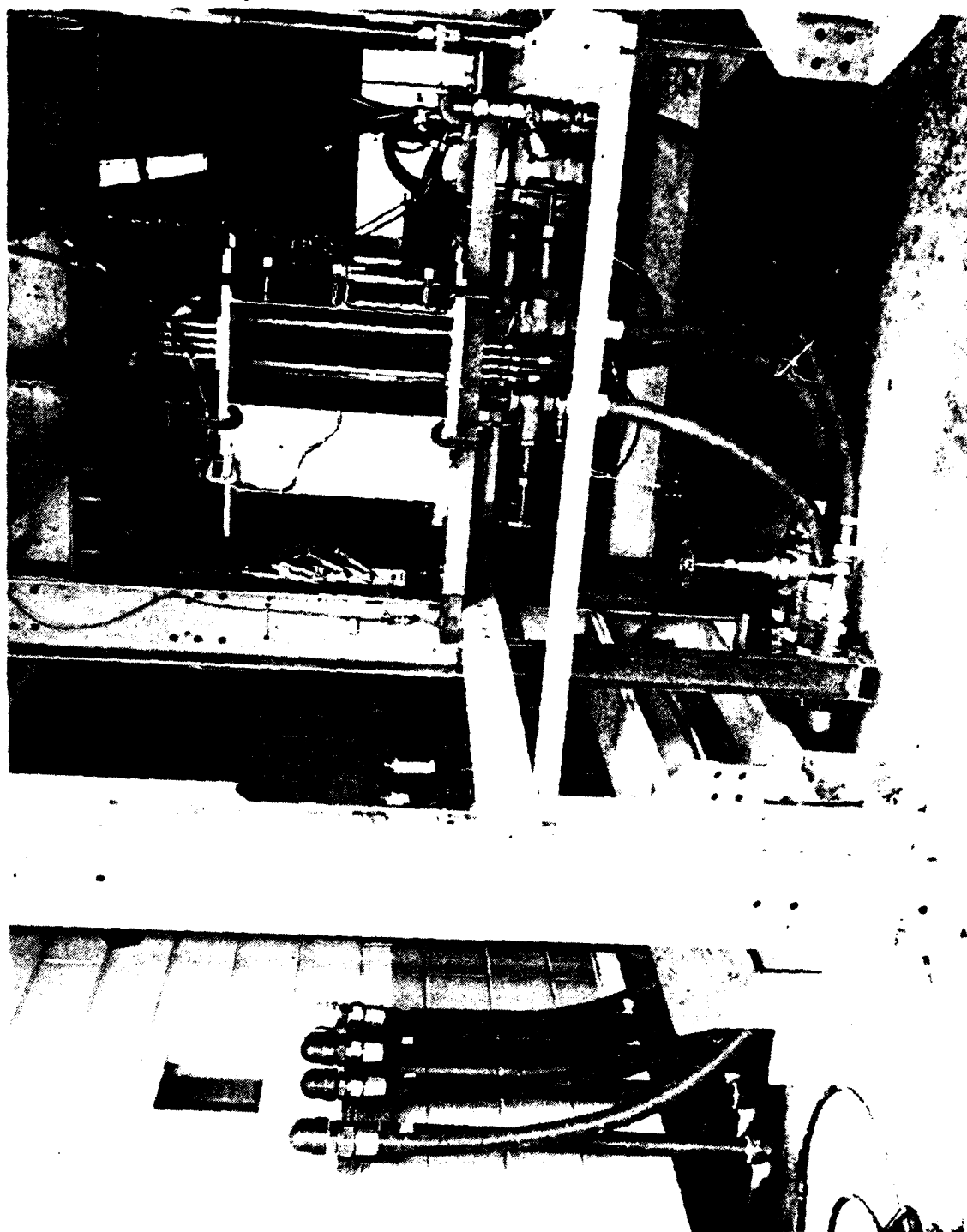
Figure 5-9. Overall View of Test Stand No. 1

(R-CAL). This was routinely done by the computer prior to each test run, and any drift in transducer excitation or signal amplification is automatically compensated for. Also prior to each test run the load cells were checked by using calibrated lead weights, and this data recorded by the computer and stored with the data for each test run.

Data acquisition and reduction is handled by an IBM 1800 computer that contains a 48 channel multiplexer and a 14 bit plus sign analog to digital converter. The computer has 32 kilowords of core storage, approximately 500 kilowords of disk storage, and a four microsecond cycle time. Input/output devices consist of a 30 character/minute keyboard printer, a 260 card/min card reader-punch, and a 240 line/minute line printer. Two analog output channels of ten bit resolution are available for plotting data on line. Seven (possible 24) levels of process interrupt control are provided to control the various functions of the processor. Two 16 bit digital output (electronic contact operated) words are available for control of external devices. One 16 bit digital input word in conjunction with an external 16 channel multiplexer is used for digital input data.

Data reduction is handled by appropriate user written computer programs. Conversion constants for all transducers are contained in the software and utilized to convert transducer electrical outputs to equivalent engineering units. Appropriate equations are programmed to calculate augmenter airflow parameters such as isentropic thrust, flow coefficients, velocity, Reynolds number, and etc. Selected parameters are calculated and punched to cards for additional data analysis by the Corporate Computing Systems TSO and CDC, which are IBM 370 and CDC Cyber 176 computers, respectively.

During the course of this test program, numerous airflow surveys, in and around the augmenter, were made with a 3-axis, computer controlled, survey rake. Both pitot-static probes and a 5-port flow angularity probe (United Sensor Mod DC-125-24-F-22-CD) were used to collect survey data. Computer programs were available to reduce pressure and position data to thrust, mass flow, velocity profile distortion, flow angularity, and etc. This type of data reduction was done on both the IBM 1800 computer as well as the Corporate Computer Systems.

Further information concerning the Thermodynamics Laboratory complex can be found in the facilities section of this document.

5.5   AUGMENTER MODELS

The augmenter models used during this test program were rectangular, 0.2 scale representations of a midspan cut of the full scale wing hardware. Rectangular models allow much quicker changes to model geometry and are considered satisfactory for evaluating the relative merits of one centerbody versus another. A cross section of a typical model is shown in Figure 5-10 along with annotations of augmenter dimensional parameters. The basic model consists of a forward and aft Coanda-flap surfaces, both with log spiral Coanda surfaces, and a centerbody nozzle. Flaps of various configurations were attached to the Coanda surfaces, and endwall

plates extending to the flap trailing edges were used to complete the
closeout of the shroud. A wing leading edge simulation was attached to
the diffuser of a length representing 0.2 scale of the full size wing.

The augmenter plenums, Coandas, flaps, endwalls, and endwall blowers were
fabricated in accordance with References (15) and (16). Two sets of
Coandas were used on Rig No. 1 during this program and were referred to as
"old Coandas" and "new Coandas." The new Coandas differ in the log spiral
shape which is defined by the initial radius (R_0) and spiral "K"
factor. In addition, the new Coandas had corner blowers incorporated at
each end of each Coanda nozzle to provide additional control of adverse
airflow in the augmenter corners.

Both Coanda shapes had the same spiral factor (K = 1.3), but the initial
radius (R_0) was changed to diffuser R_0 = .483 inches and elevon
R_0 = .4156 inches. These R_0's gave R_0/t values of 10.39 and 12.075,
respectively, for the diffuser and elevon, and reflect the aircraft values
for a midspan cut, while the "old Coandas" had an R_0/t = 8.7 for
diffuser and elevon, which more nearly represented aircraft values at the
outboard end of the nozzles.

Centerbody performance testing using the "old Coandas" was done with the
7°, 15°, 22.5°, and 28° hypermixing nozzles, as well as with the 14
element asymmetric, 14 element asymmetric mods 1 and 2, and 18 element
asymmetric. All subsequent asymmetrics and mods as well as all symmetric
centerbody nozzles were tested using the "new Coandas." All the
hypermixing centerbodies plus the 14 element asymmetric mod 2 were also
retested with the "new Coandas" for a comparison data base.

Details of the Coanda nozzle internal construction can be seen in
Figure 5-11. The Coanda surface is integral with the internal radius
surface and attached to the plenum tube which also forms the nozzle lip.
The attachment is made utilizing a slotted hole which allows nozzle gap
adjustments.

Air was supplied to the plenum through a feed pipe which has a row of
holes the full length of the nozzle, exiting air into the plenum in a
direction opposite to the nozzle exit. Spanwise pressure distribution was
accomplished by enlarging or plugging selected holes as required. The
ratio of plenum volume to nozzle exit area was sufficient to uniformly
choke the nozzle with a minimum of spanwise flow.

The inboard ends of the plenum tubes were fabricated with two grooves on
the outside diameter; one groove provided sealing utilizing an O-ring and
the other was a recess for set screws which held the nozzle to an adapter
fitted to the inboard endwall. This arrangement allowed the rotation of
the nozzles within the adapter, thereby permitting variations in Coanda
turning angle, that is flow angle from nozzle exit to throat.

The corner blowers (dedicated nozzles) shown in Figure 5-11, located at
each end of each Coanda nozzle were supplied with air from the same plenum
as the primary nozzle. Convergence to the nozzle exit was provided to
ensure a uniform exit pressure profile.

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	K	R_0	R_0/t
OLD DIFF	1.3	.348	8.70
NEW DIFF	1.3	.483	10.39
OLD ELEVON	1.3	.348	8.70
NEW ELEVON	1.3	.4156	12.075

	TAP NO	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
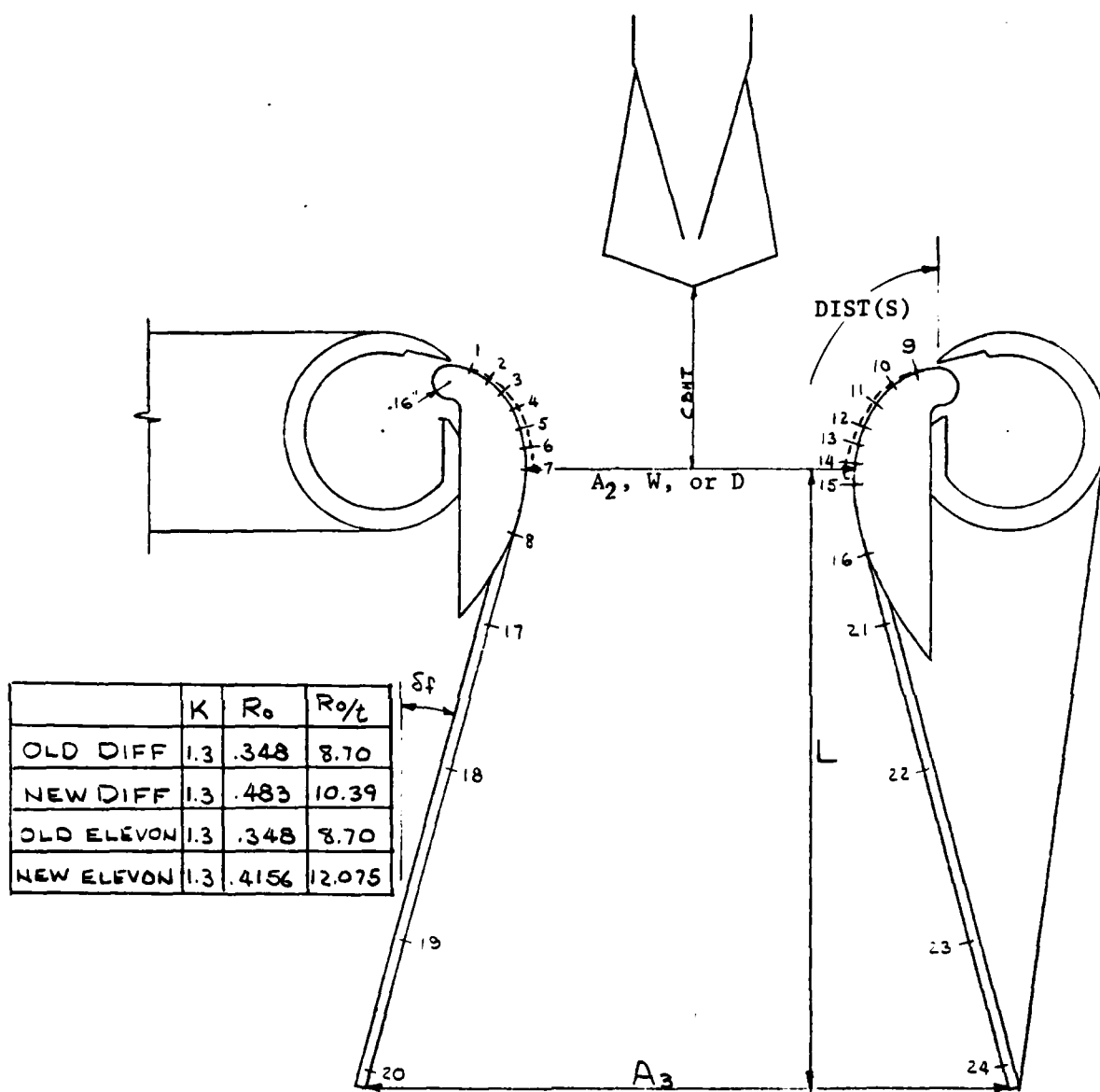
PS TAP NO & COANDA SURFACE DISTANCE

NOZZLE EXIT-TO-TAP

	TAP NO	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
RIG 1	DIST	.17	.37	.59	.73	.91	1.09	1.28	1.88	.24	.48	.77	.94	1.14	1.31	1.50	2.13
RIG 5	DIST	.25	.43	.60	.77	.96	1.13	1.34	1.92	.18	.42	.67	.90	1.10	1.26	1.46	2.12

TAP LOCATIONS TESTS 278, 294, 304

Figure 5-10. Augmenter Geometry

Rockwell International
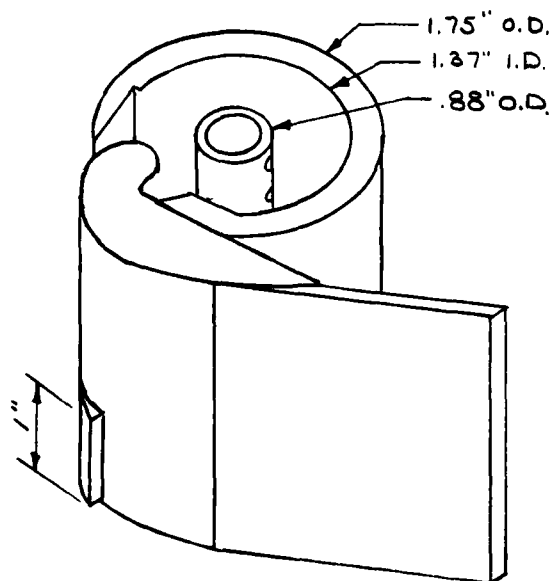
COANDA NOZZLE - FLAP
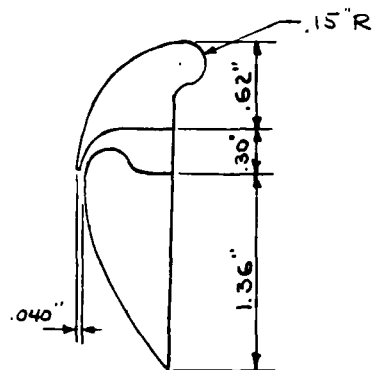
CORNER BLOWER
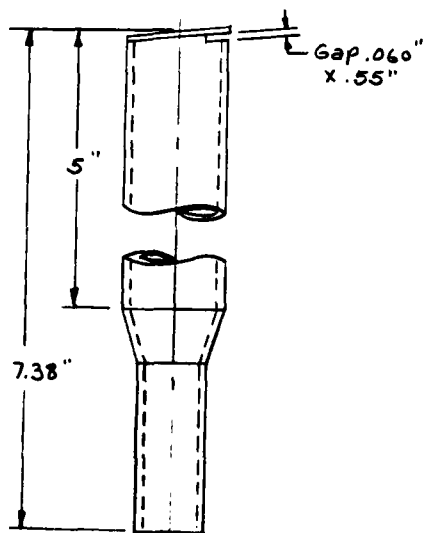
NICKEL BLOWER
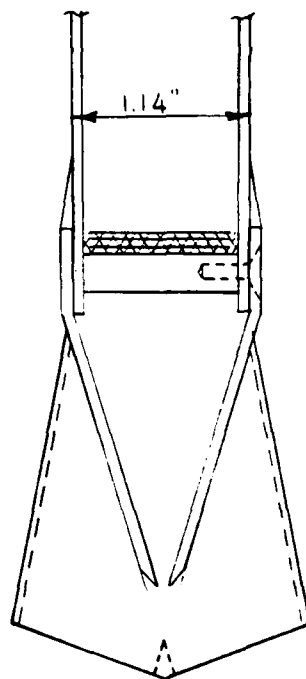
CENTERBODY NOZZLE

Figure 5-11.   Augmenter Model Construction Details

Figure 5-10 shows locations of static taps on the Coanda surfaces which were located at midspan of the nozzle. Pressure taps 1 through 16 were on the actual Coanda surfaces while taps 17 through 24 were on the flaps. The table in Figure 5-10 provides tap locations as distance on the surface from the nozzle exit. Location of taps 17 through 24 varied with the particular flap used, and this information may be obtained from Figures 5-12, 5-13, and 5-14 for each individual flap.

Starting with a 16 April 1981 date, additional surface static pressure taps were added to the model on test stand No. 5. These taps were added to the Coanda-flap surfaces as well as to the upper surface of the wing leading edge simulation. A new test number (307) was assigned to the Phi runs of the rig 5 model to mark this major change. Figure 5-15 shows the locations of both the original and added static pressure taps.

The method of attaching centerbody nozzles to the centerbody plenum is shown in Figure 5-11. This type of construction resulted in cost and time savings as nozzle changes could be made by fabricating only the nozzle element section and utilizing a common plenum. This type of centerbody nozzle construction also resulted in time savings with regard to obtaining uniform nozzle pressure distributions, in that the screening, used to distribute airflow uniformly along the span of the nozzle, remained with the plenum and did not require changing when nozzle elements were changed, so long as the nozzle exit areas were approximately the same.

The effect of internal screening of a nozzle on the jet turbulence intensity was studied using a hot film anemomenter. It was determined, the screening had little effect. Figure 5-16 presents the turbulence intensity profiles for three nozzle screening configurations and Figure 5-17 presents a ϕ versus A_3/A_2 comparison of two screening configurations.

Boundary layer control devices (BLCS) were incorporated in the endwalls located at or slightly below the augmenter throat line (see Figure 5-18) in the endwalls to provide control of airflow separation on these surfaces. The BLCS consisted of three or four (depending on throat dimension) rotatable slot nozzles exiting air parallel to the endwalls at a nozzle pressure ratio comparable to that of the other augmenter nozzles.

The amount of air used in the BLCS was typically four to five percent of the total exit area (A_0) of the augmenter model. Figure 5-11 shows a sketch of a typical endwall "nickel blower" nozzle.

The inboard and outboard endwalls (see Figure 5-19) were fabricated in three sections which allowed the flexibility of making augmenter throat variations without a major change to the model. Spaces resulting between the sections were easily filled with spacer blocks and smoothed with epoxy compounds. A similar arrangement was utilized to provide movement of the centerbody so that centerbody height sensitivities could be studied.

15° 5.94" FLAPERON (356-01042-191)
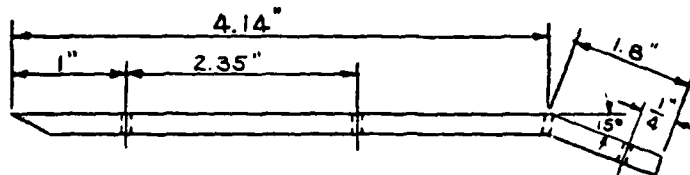
10° 6.3" FLAPERON (356-01042-251)
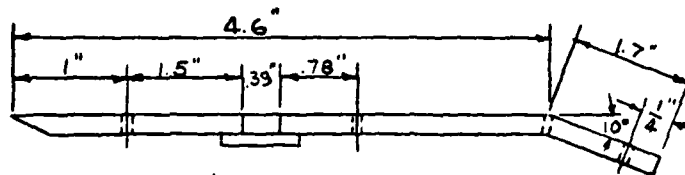
15° 6.3" FLAPERON (356-01042-261)
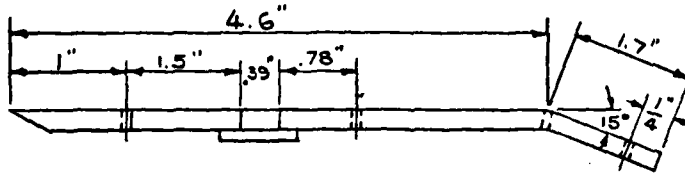
15° 6.3" FLAPERON (356-01042-271)
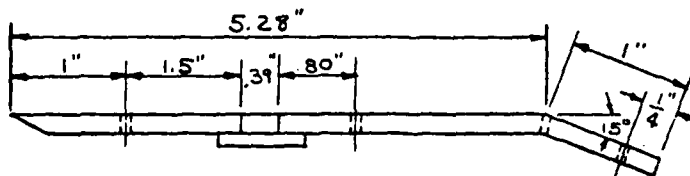
12.3" STRAIGHT FLAP DIFFUSER (ELEVON)
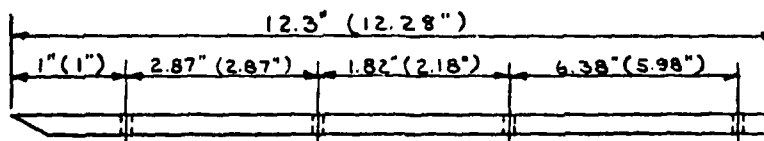
Figure 5-12. Model Flap Pressure Tap Locations

7" STRT FLAP (356-01042-063)
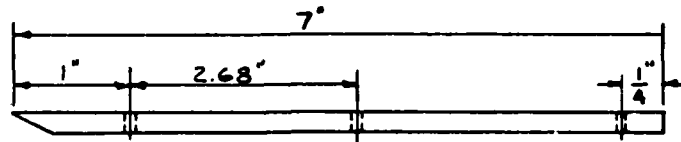
10° 6" FLAPERON (356-01042-151)
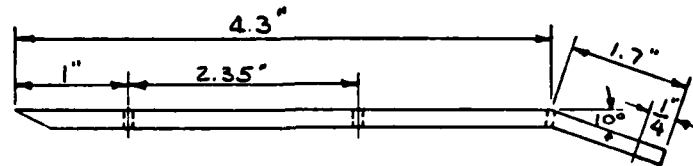
15° 6" FLAPERON (356-01042-161)
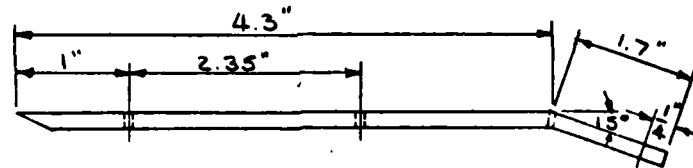
15° 6" FLAPERON (356-01042-171)
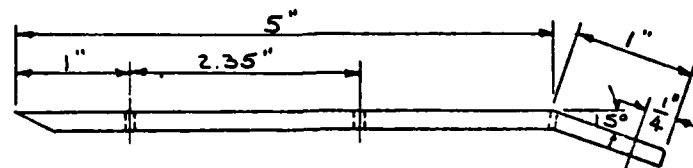
15° 7" FLAPERON (356-01042-181)

Figure 5-13. Model Flap Pressure Tap Locations

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6.3" HINGED FLAP (356-01042-201)
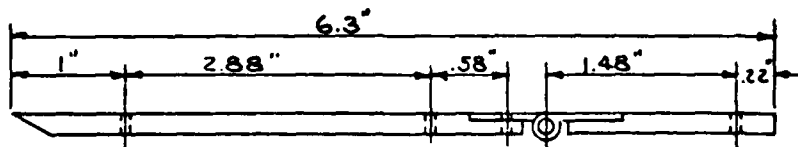
5.94" HINGED FLAP (356-01042-301)
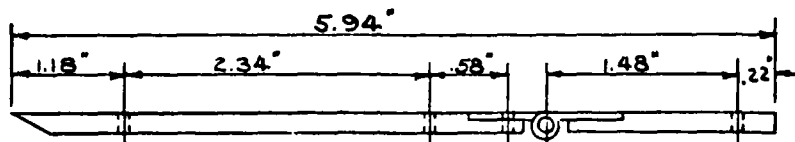
6.32" STRAIGHT FLAP (356-01042-251/257)
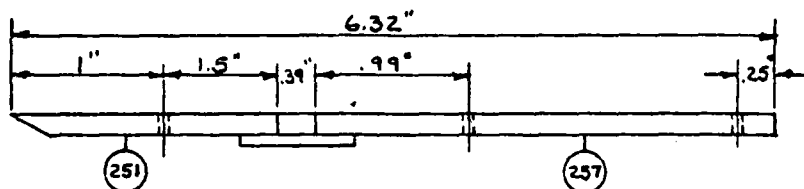
5.95" STRAIGHT FLAP (356-01042-251/259)
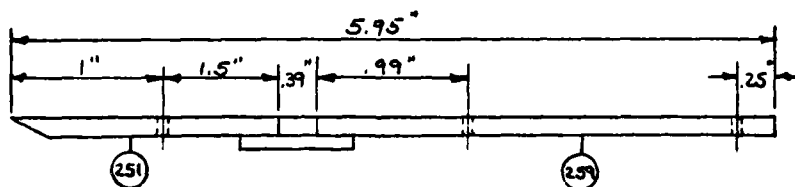
9.5" STRAIGHT FLAP DIFFUSER (ELEVON)
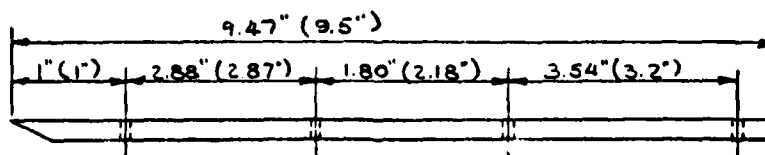
Figure 5-14. Model Flap Pressure Tap Locations
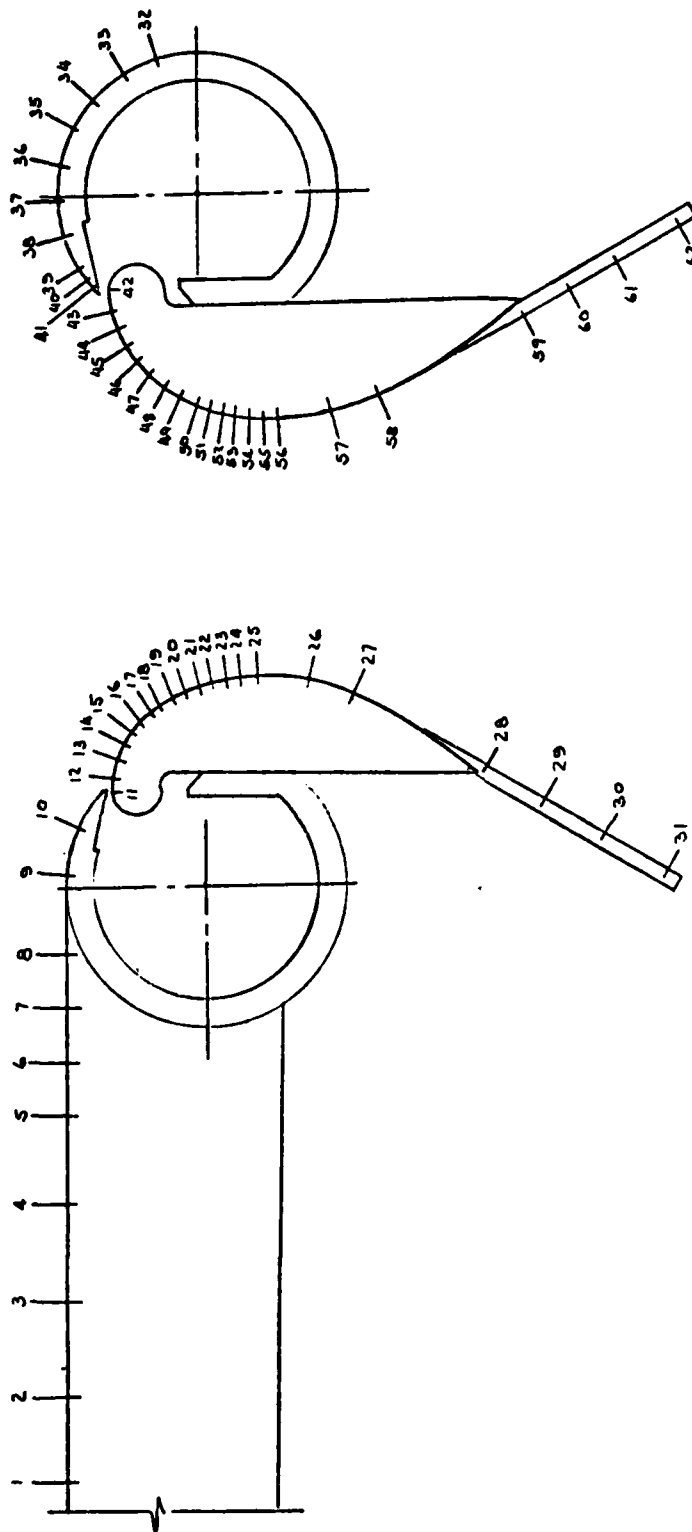
REF No vs SCV MODULE No/PORT No vs DIST FROM NOZZ EXIT

REF No	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30	31
MODULE/PORT No	1/37	1/36	1/35	1/34	1/33	1/32	1/31	1/30	2/3	2/4	2/5	2/6	2/7	2/8	2/9	2/10	2/11	2/12	2/13	2/14	2/15	2/16	2/17	2/18	2/19	2/20	2/21	1/2	1/3	1/4	1/5
DIST	6.70	5.69	4.69	3.67	2.67	2.18	1.69	1.17	.63	.28	.04	.14	.25	.36	.43	.53	.61	.68	.78	.88	.96	1.06	1.15	1.26	1.34	1.47	1.95				

REF No	32	33	34	35	36	37	38	39	40	41	42	43	44	45	46	47	48	49	50	51	52	53	54	55	56	57	58	59	60	61	62
MODULE/PORT No	1/29	1/28	1/27	1/26	1/25	1/24	1/23	1/22	1/26	2/21	2/24	2/30	2/31	2/31	2/32	2/33	2/33	2/34	2/35	2/34	2/37	2/38	2/39	2/40	2/41	2/42	2/43	1/6	1/7	1/8	1/4
DIST	1.94	1.44	1.21	.96	.73	.49	.25	.25	.09	.04	.10	.18	.25	.32	.43	.55	.63	.80	.91	1.00	1.11	1.17	1.27	1.36	1.46	1.76	2.13				

Figure 5-15. Surface Static Press Tap Location ~ Rig 5, Test 307

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EFFECT OF INTERNAL SCREENING
ON JET TURBULENCE INTENSITY

SCBT15-1 CENTERBODY ~ NOZZLE VEL ≅ 900 fps
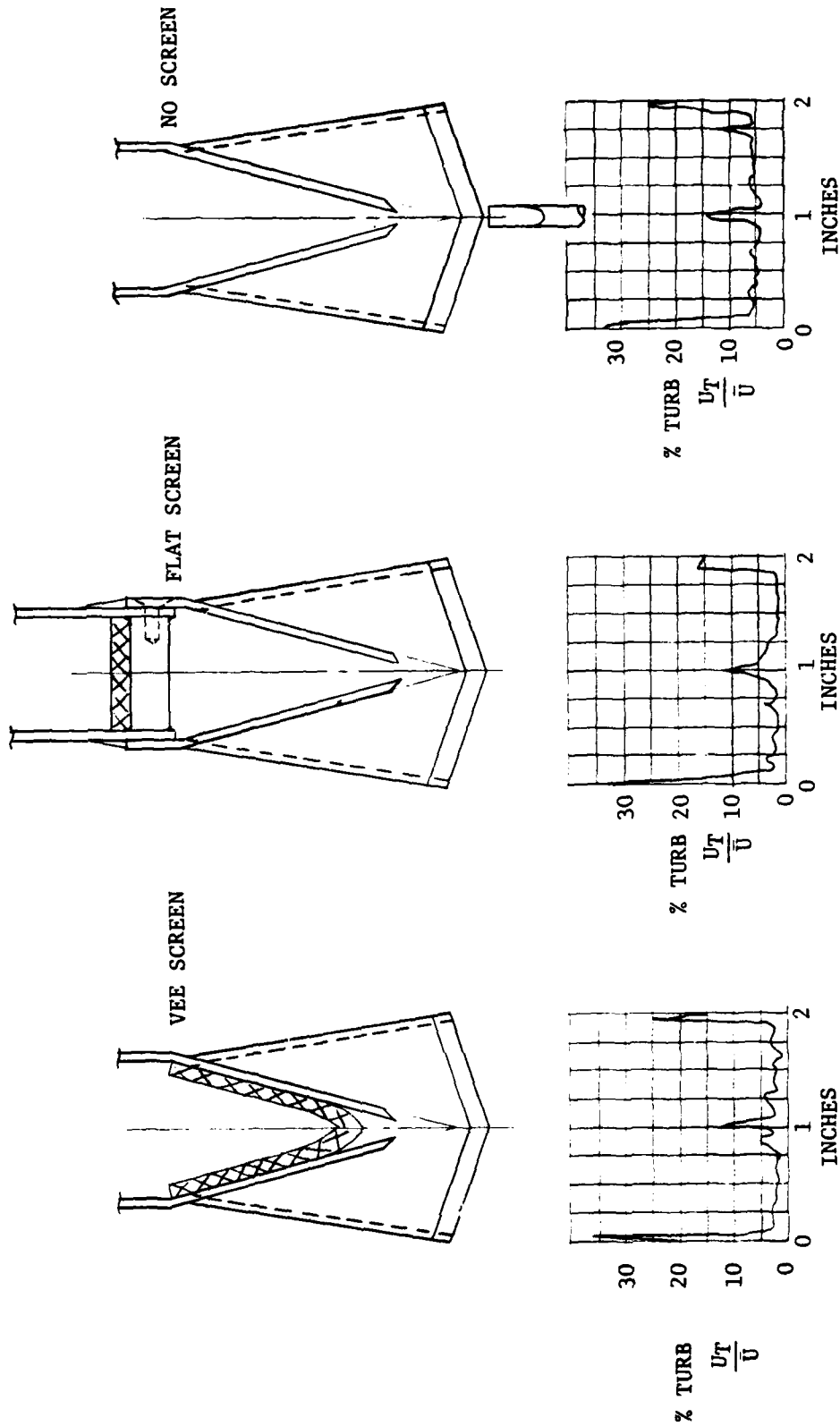
Figure 5-16.
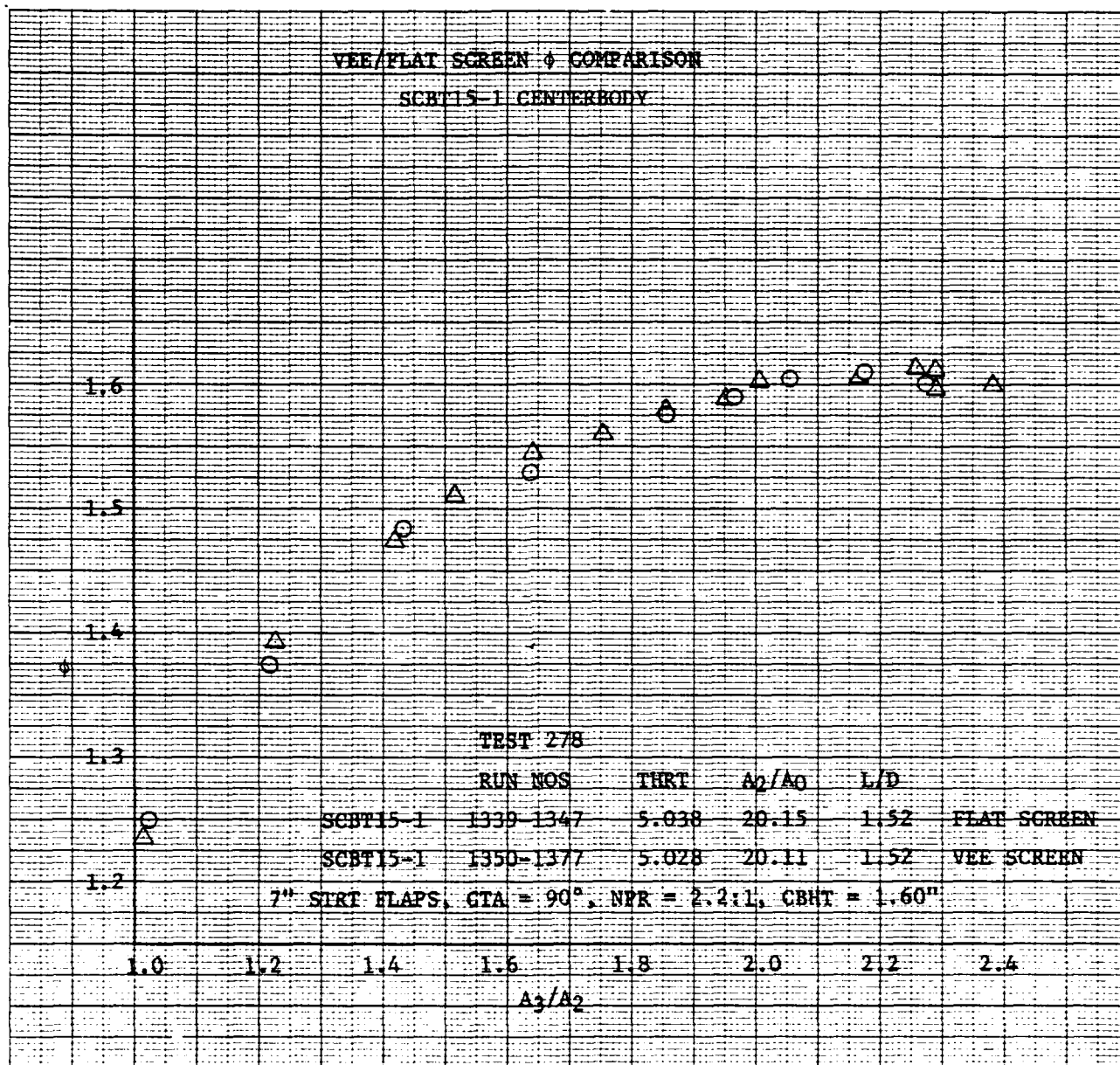
Figure 5-17   Vee/Flat Screen ϕ Comparison SCBT15-1 Centerbody

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ENDWALL BLOWER LOCATIONS

THT. = 5.029

G   F

A — B — C — D — E

RIG 1

THT. = 5.028

G   F

A — B — C — D — E

RIG 5

THT. = 3.24

F   G

A — B — D — E
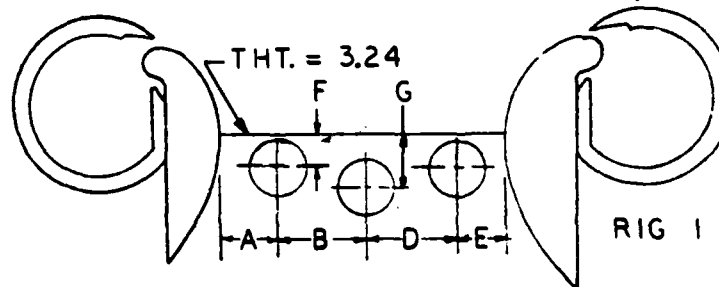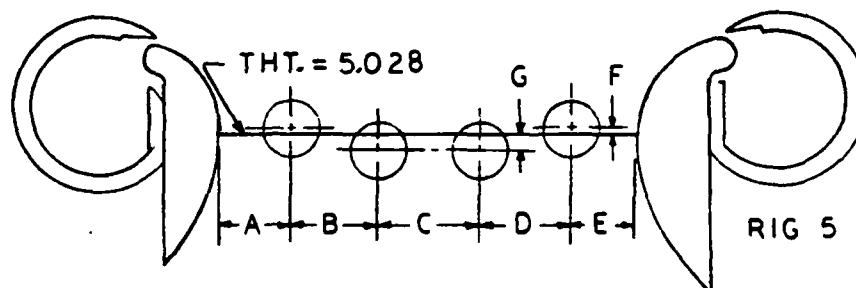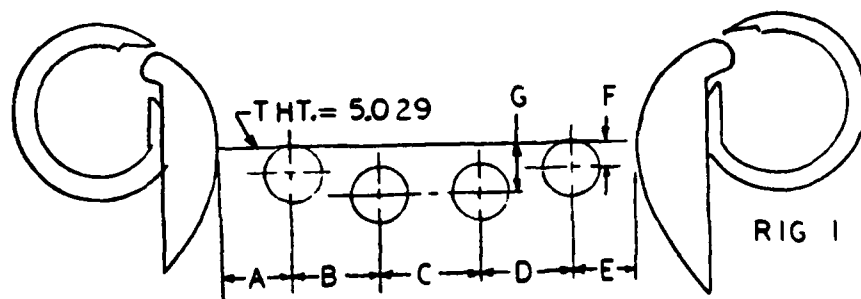
RIG 1

	A	B	C	D	E	F	G
RIG 1 IB	1.05	.98	.98	.97	1.05	.36	.59
RIG 1 OB	1.10	.98	.98	.97	1.00	.35	.55
RIG 5 IB	1.26	.79	.95	.80	1.22	.19	.35
RIG 5 OB	1.25	.90	.90	.75	1.22	.19	.35
RIG 1 IB/OB	.75	.87	—	.87	.75	.50	.75

NOTE: 1. SKETCH NOT TO SCALE.
2. DIMS. A & E WILL VARY WITH THE THROAT. DIMS.
SHOWN ARE FOR THROATS INDICATED.

Figure 5-18   Endwall Blower Locations
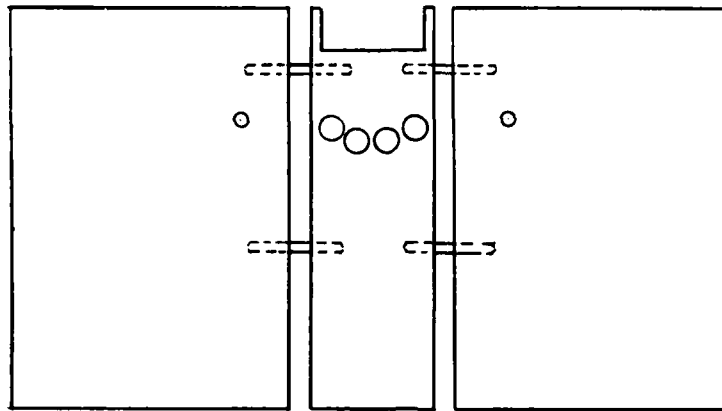
OUTBOARD ENDWALL
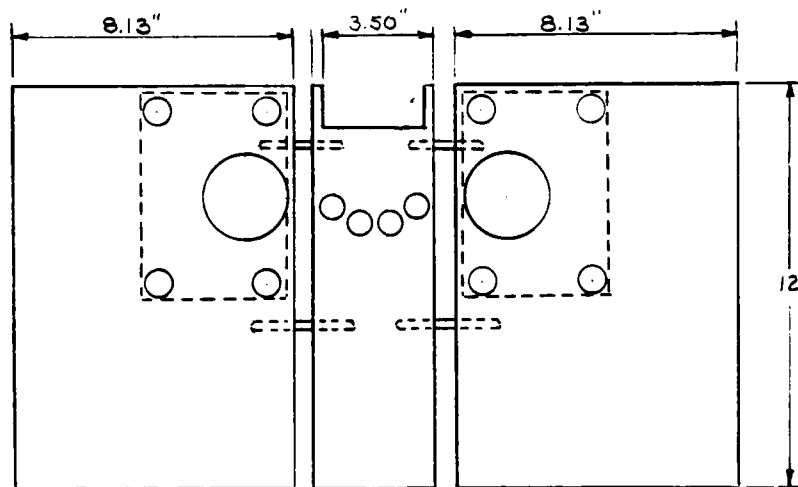
INBOARD ENDWALL

Figure 5-19 Augmenter Endwall Construction Details

5.6   INSTRUMENTATION ACCURACY

Since the measured ϕ is defined by the equation $\phi = L/\dot{m}V$ where

Lift = Measured load stand thrust
$\dot{m}$ = Measured venturi mass flow
V = Nozzle exit velocity based on nozzle pressure ratio.

The fractional error in ϕ may be defined as

$$\frac{\Delta\phi}{\phi} = \sqrt{\left(\frac{\Delta L}{L}\right)^2 + \left(\frac{\Delta\dot{m}}{\dot{m}}\right) + \left(\frac{\Delta V}{V}\right)^2} \quad .$$

Load Cell Error - The lift load cells are calibrated using dead weights which are calibrated on a HOMS scale having a calibration accuracy of $\pm$.11 percent. The lift load cells mounted in the test stand gave an average error of .224 percent based on 31 separate test runs. Tolerance on hose tare corrections was found to be .006 percent. The total expected probable error in lift measurements was calculated to be .25 percent.

Venturi Meter - The Venturi meters used during this test program have a basic accuracy (C_D) of $\pm$ 1.0 percent based on calibrations supplied by the Mechanical Engineering Department of The Ohio State University.

Pressure Transducers - The three pressure transducers used in determining ϕ were an upstream and ΔP transducer on the Venturi meter used to calcuate mass flow, and a transducer to determine nozzle pressure ratio which is utilized to calculate jet velocity. The working standard to calibrate transducers has a basic accuracy of $\pm$.06 percent. The transducers have accuracy limits of

- Hysteresis $\pm$.10 Percent Full Scale
- Non-Linearity $\pm$.25 Percent Full Scale
- Repeatability $\pm$.050 Percent Full Scale
- Temperature on Span $\pm$.005 Percent
- Temperature on Zero $\pm$.005 Percent

The readout accuracy of pressures through the data system to the computer are estimated to be .061 percent.

Combining the above accuracy limits gives a total probable error in pressure measurements of $\pm$.28 percent.

$$\frac{\Delta P}{P} = \sqrt{\begin{array}{l}(.0006)^2 + (.0010)^2 + (.0025)^2 + (.0005)^2 + (.00005)^2 \\ + (.00005)^2 + (.00061)^2\end{array}} = .0028$$

Velocity - The equation for jet velocity used in the data reduction programs was:

$$Vel = \sqrt{\frac{\gamma \cdot R \cdot Temp \cdot 2\left(1 - NPR^{\frac{1-\gamma}{\gamma}}\right)}{\gamma - 1}}$$

where $\gamma = 1.4$, $R = 1716$, Temp = Air Temp in °R, NPR = nozzle pressure ratio. Assuming an error in pressure measurement of .28 percent, an error in jet velocity of .117 percent can be expected.

Mass Flow - The equation for mass flow used in the data reduction programs is:

$$\dot{m} = \frac{Ya \cdot \pi \cdot Dv^2 \cdot C_D \sqrt{2 \, g^2 \cdot (P_{up}/(Temp \cdot R)) \cdot \Delta P}}{4\sqrt{1 - Av^4}}$$

$$\text{where } Ya = \left[\frac{\left(PR^{\frac{2}{\gamma}}\right)\left(\frac{\gamma}{\gamma-1}\right)\left(1 - PR^{\frac{\gamma-1}{\gamma}}\right)(1 - Av^4)}{(1 - PR)\,(1 - (Av^4 \cdot PR^{2/\gamma}))}\right]^{\frac{1}{2}}$$

Dv	=	Venturi throat dia. inches
C_D	=	Venturi flow coefficient = .99 $\pm$ 1%
g	=	32.2 ft/sec²
P_{up}	=	Vent upstream pressure
Temp	=	Vent Air Temp °R
R	=	1716 ft²/sec² °R $\left(53.35\left(\frac{ft \cdot lbs_f}{°R \, lbs_m}\right) \times 32.2\left(\frac{lbs_m \cdot ft}{lbs_f \, sec^2}\right)\right)$
ΔP	=	Venturi upstream press - Venturi throat press
Av	=	Venturi throat dia./Venturi upstream dia.
PR	=	Venturi throat press/Venturi upstream press

Assuming an error of .28 percent in the upstream and ΔP pressure measurements and a one percent error in Venturi flow coefficient, a probable error in mass flow measurements of 1.29 percent can be expected.

Combining the above errors in lift, velocity, and mass flow:

$$\frac{\Delta \phi}{\phi} = \sqrt{(.0023)^2 + (.00117)^2 + (.0129)^2} = .0132$$

An expected probable error in 0 due to instrumentation errors is therefore approximately 1.3 percent. It will be noted that in calculation of

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velocity and mass flow no error was assumed to be made in measurement of temperature. Examining the equations for ṁ and Velocity it can be seen that the product of these two equations will result in temperature effects cancelling.

5.7 TEST PROCEDURES AND METHODS

5.7.1 Short Time Repeatibility

To determine the possible variation in augmenter performance over a short period of time and the possible error encountered by taking lift and pressure data at an instantaneous time slice, a series of runs were made comparing results where ten data samples were taken and averaged and compared to results of a single sample data point. Figure No. 5-20 is a comparison plot of single sample/data point versus ten samples/per data point at one sec Δ between samples. Results show that ϕ at any given time is comparable to ϕ over a period of ten secs.

This sampling/average technique was further tested by increasing the time between samples from one sec. to ten sec. to spread the data acquisition per data point over a period of 100 secs. The lower plot on Figure No. 5-20 shows the results of one sample/sec. versus one sample/ten sec for a ten sample data point. Comparing the standard deviation for these two runs at 2.2:1 NPR shows the one sample/sec. data point and one sample/ten sec data point have standard deviations of .0026 and .0019, respectively. Based on these studies, the one sample per second – ten sample data point technique was incorporated as a permanent part of the data acquisition program, starting on April 8, 1980.

5.7.2 Long Time Repeatability

During a series of test runs where changes to the model's flaps and endwalls were being made, it was thought prudent to restore the model to a baseline configuration periodically to check the repeatability of the model's performance. Figure 5-21 shows a ϕ versus A_3/A_2 comparison of four run series over a two week period, where the maximum scatter in ϕ performance was approximately $\pm.013$. This sort of repeatability check was made periodically throughout the program, wherever an opportunity presented itself. On occasion, a centerbody that had previously been tested, was reinstalled in the model to obtain additional data, and this gave the opportunity to make a check run to insure no undetected sources of error had developed in the model or instrumentation.

5.7.3 Rig 1 to Rig 5 Data Comparison/Repeatability

During the nozzle development program, a second test Rig (No. 5) was brought into use and an augmenter shroud fabricated to the same specifications as the augmenter on Rig No. 1.

In order to test centerbodies on both rigs and have confidence that the data could be accurately compared, test runs were made on Rig 1 with a centerbody/shroud configuration that had previously been tested on Rig
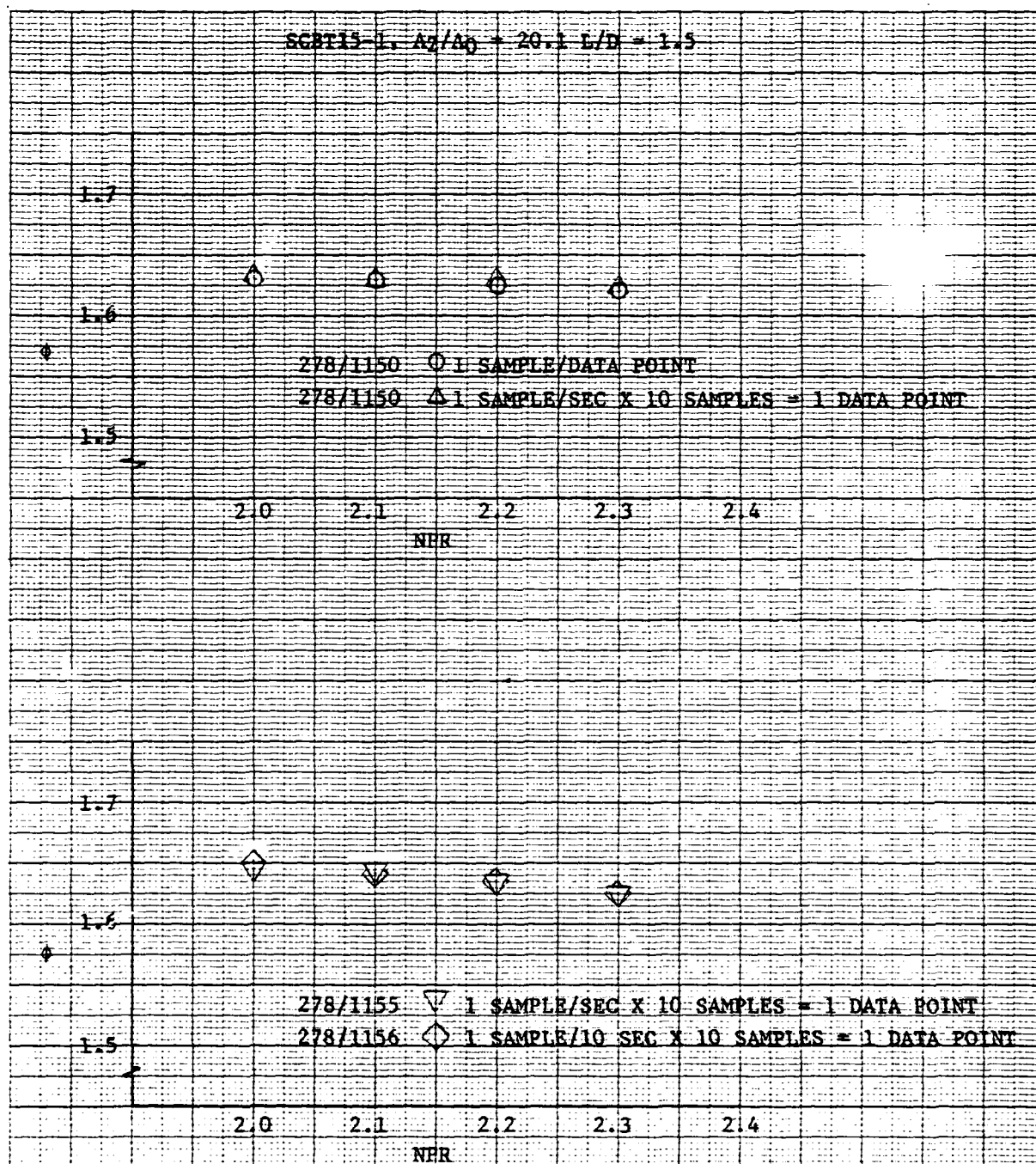
Figure 5-20 Data Repeatability - Short Term ϕ Versus NPR

SCATTE5-9 CENTERBODY
$A_2/A_0 = 19.9$ $L/D = 1.5$ NPR $= 2.2:1$

◯ 1ST SERIES RUNS (274-292)
◻ 2ND SERIES RUNS (320-328)
◇ 3RD SERIES RUNS (337-341)
▽ 4th SERIES RUNS (342-346)

2.4 2.2 2.0 1.8 1.6 1.4 1.2 1.0

A_3/A_2

1.7 1.6 1.5 1.4 1.3 1.2

Φ

Figure 5-21. DATA REPEATABILITY - LONG TERM.
Φ VERSUS A_3/A_2
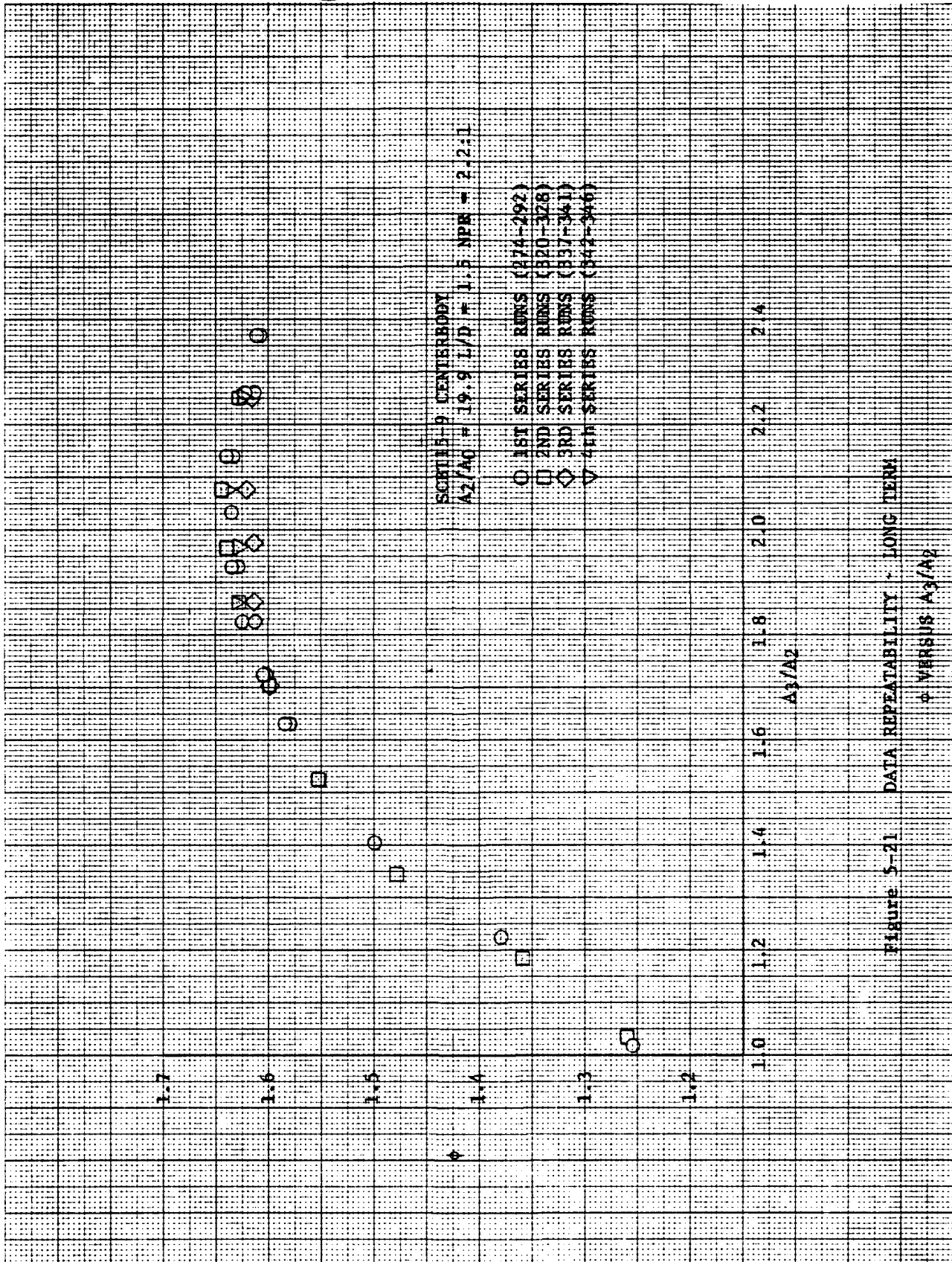
No. 5. Figure 5-22 shows a comparison plot of the ϕ's obtained for two centerbody nozzles on Rigs 1 and 5. Results were quite comparable and allowed centerbodies to be tested on either rig without possible rig or shroud differences clouding the data.

5.8 SUMMARY

Analytical methods for predicting the entrainment of the turbulent jets and pressure distributions on the ejector flap have been described. Consideration of physical laws, mathematical analysis, and experimental data have been presented to show that scale model ejectors, powered by cold air jets, can be used to aid in the development of full size ejectors, powered by hot exhaust gases. Experimental apparatus and test procedures devised to ensure accurate and repeatable data are also described.

RIG 1/RIG 5 COMPARISON

SCBT15-1 & OLD BT5 CENTERBODIES

	RUN NOS	THRT	A_2/A_0	L/D
SCBT15-1 — RIG 1	1339-1347	5.038	20.15	1.51
SCBT15-1 — RIG 5	126- 137	5.026	19.98	1.49

7" STRT FLAPS, CTA = 90°, NPR = 2.2:1, CBHT = 1.60"

A_3/A_2

	RUN NOS	THRT	A_2/A_0	L/D
OLD BT3 — RIG 1	435-446	4.490	17.63	1.65
OLD BT3 — RIG 5	6- 14	4.498	17.75	1.66

7" STRT FLAPS, CTA = 90°, NPR = 2.0:1, CBHT = 1.60"

A_3/A_2

Figure 5-22 Rig 1/Rig 5 Comparison
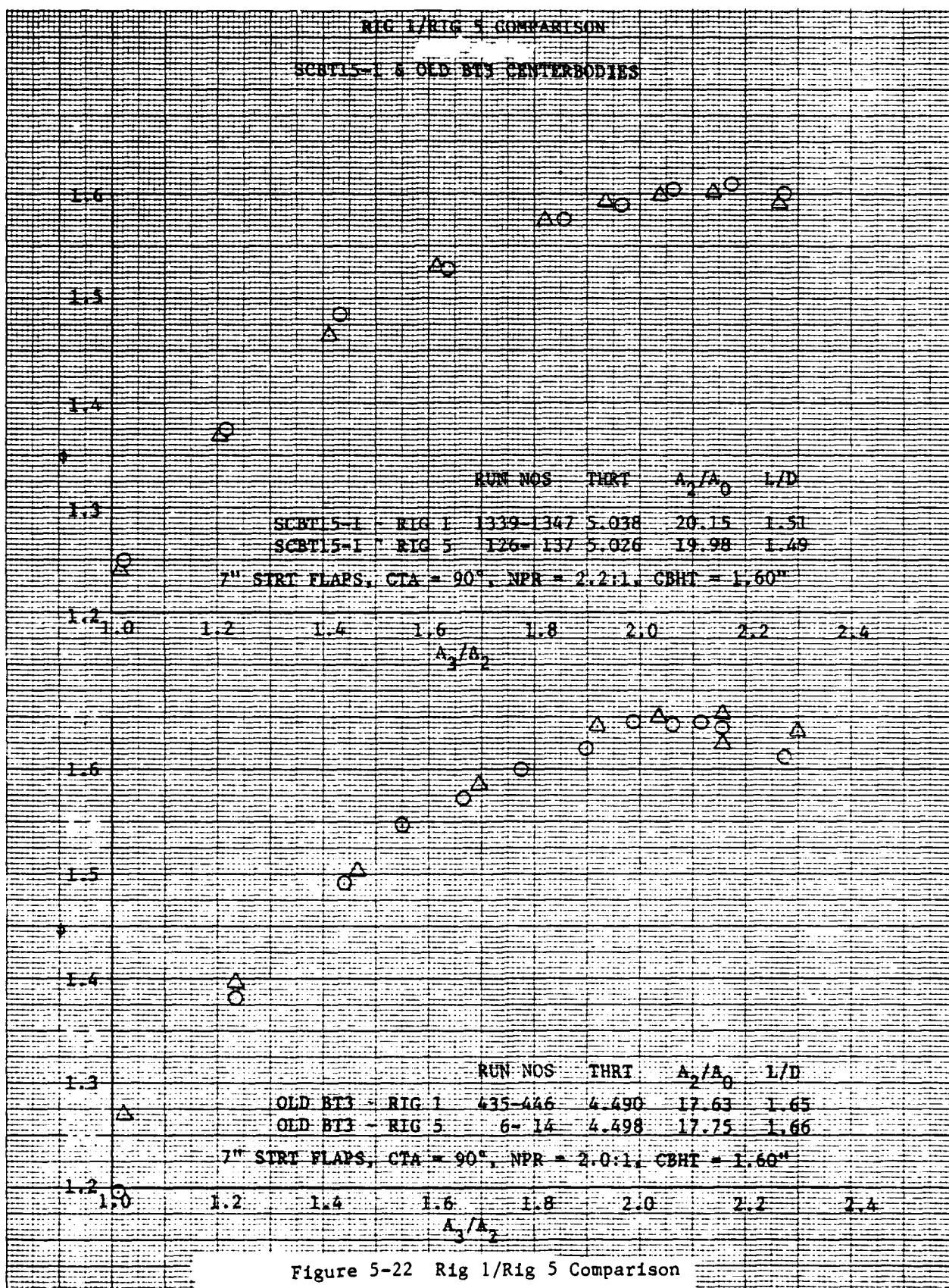
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6.0 COANDA DEVELOPMENT

6.1 INTRODUCTION

As a result of the XFV-12A wing diagnostic test program discussed in Section 2.3.6 of this report, several flow deficiencies associated with the wing Coanda-flap components were identified. As discussed previously in Section 3.1, an improvement rationale was selected for each of the major augmenter components. For the Coanda-flap surfaces, this approach entailed utilizing both model scale and full scale component isolated testing in combination with aircraft design and packaging studies. Based on these investigations, Coanda selection criteria were developed which identifed candidate configurations satisfying both performance and design criteria. These candidate configurations were then evaluated on a full scale segment model simulating aircraft feed characteristics. Due to the unique flexibility of this model, additional studies evaluating the candidate configuration's sensitivity to inlet Mach number, feed duct position, internal Coanda shapes, and various internal flow turning devices were also performed. Finally, the optimum external Coanda shape in combination with other selected geometric modifications were incorporated and evaluated on the L/H XFV-12A wing aft Coanda-flap component (elevon).

This data when compared to comparable data obtained from the baseline XFV-12A elevon provided an indication of the relative improvement and Coanda flow quality and performance which resulted from the Coanda development effort.

6.2 SCALE COANDA EVALUATION

During the course of augmenter development work, a number of Coanda shapes were evaluated to identify a shape which would maximize augmenter performance and at the same time be compatible with aircraft packaging constraints. A number of factors must be considered when selecting a specific shape, to minimize losses and prevent jet separation from the surface while the jet is being turned. It has been determined empirically, for the nozzle pressure ratios currently being tested, a radius of curvature (R_0) at the nozzle exit must be at least approximately eight times the nozzle gap (t) to prevent separation at the nozzle. In addition, to prevent jet separation downstream of the nozzle exit, while the Coanda surface is operated in an augmenter, a "rule of thumb" has been observed that the local radius of curvature should be at least four times the local jet width ($R_\theta \geq 4$). Also the radius of curvature at the point where the Coanda surface joins the straight flap should be as large as possible to prevent a sudden diffusion loss as the jet passes the point where the radius changes from some value (however large) to a radius of infinity for the straight flap.

Another consideration while turning the jet is the running length, where a long running length (s) would maximize the distance for entrainment of the co-flowing stream, but would also increase the surface friction losses.

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Packaging considerations provide another constraint on the Coanda shape in that the desirable larger radii (R_i and R_o) result in thicker nozzles which are difficult to enclose in a relatively thin supersonic wing section. A convenient shape easily definable mathematically, allows specifying R_o at the nozzle exit and provides for a gradually increasing radius of curvature is the log spiral shape. Figure 6-1 presents the log spiral and the equations which define its shape and some of its characteristics. Also shown is this log spiral integrated into a Coanda nozzle where it can be seen which spiral parameters are determined by the maximum allowable space available for packaging the nozzle. The dimension Y will increase in proportion to $R_{90°}$ which in turn increases with an increase in R_o and a decrease in the sprial K factor. In the present forward and aft flap design, the tangent point of the flap with the log spiral occurs at $\theta = 106°$ and it is desirable to have this parameter ($R_{\theta=106°}$) as large as possible.

The internal radius of curvature (R_i) should also be as large as possible for two reasons: (1) to provide as gradual as possible convergence towards the nozzle exit so as not to distort the pressure profile at the exit, and (2) to provide as gradual a change as possible between R_i and R_o at the nozzle exit to prevent the sudden diffusion loss associated with a sudden change in radius of curvature mentioned above. However, large R_i's also demand that the X dimension increase, which in turn drives the total surface thickness upward.

A number of 0.2 scale log spiral Coanda shapes were evaluated in free air to determine the performance characteristics of Coanda nozzles that might be acceptable. Table 6-1 presents the nozzles tested and some of their physical parameters.

Performance characteristics measured for these nozzles were: jet spreading, nozzle efficiency in the form of thrust coefficient (C_T), and Coanda Turning Efficiency (CTE), which is a measure of the pressure remaining in the jet after being turned. The performance characteristics were measured at Coanda turning angles (CTA) of 30°, 60°, 90°, and 120° at nozzle pressure ratios (NPR) from 1.5:1 to 3.25:1. References (17) and (18) document the results of these tests in detail and some of the results are presented here. Table 6-2 lists the spiral shapes tested and compares their performance.

The entrainment index (I) discussed in Reference (17) is essentially the product of local jet width (w) normalized to nozzle gap (t) and local peak jet velocity (V_p) normalized to nozzle avg. exit velocity (V_o). Some observations of the flow quality of these Coanda surfaces are listed below according to Coanda spiral number.

No. 1   Marginally acceptable - lower than average C_V and I - Large R_i/t gives thick nozzle.

No. 2   Good performance - Highest entrainment index (I) and better than average C_V - However, large jet width to local radius ratio (W_{90}/R_{90}) - likely to separate when operated in an augmenter.

$$R_\theta = R_0 \, e^{\theta/K} \qquad \phi = TAN^{-1} \frac{1}{K}$$

$$P_\theta = P_0 \, e^{\theta/K} \qquad S_\theta = R_0 K \left(e^{\theta/K} - 1 \right)$$

$$P_0 = R_0 \cos \phi$$

$$= \frac{R_0}{\left(1 + \frac{1}{K^2}\right)^{\frac{1}{2}}} \qquad Y = \frac{R_0 \left[e^{\pi/2K} - \frac{1}{K} \right]}{1 + \frac{1}{K^2}}$$
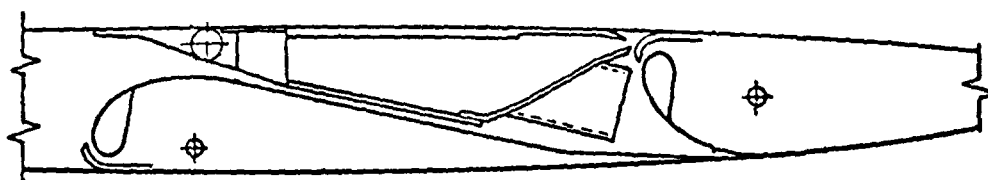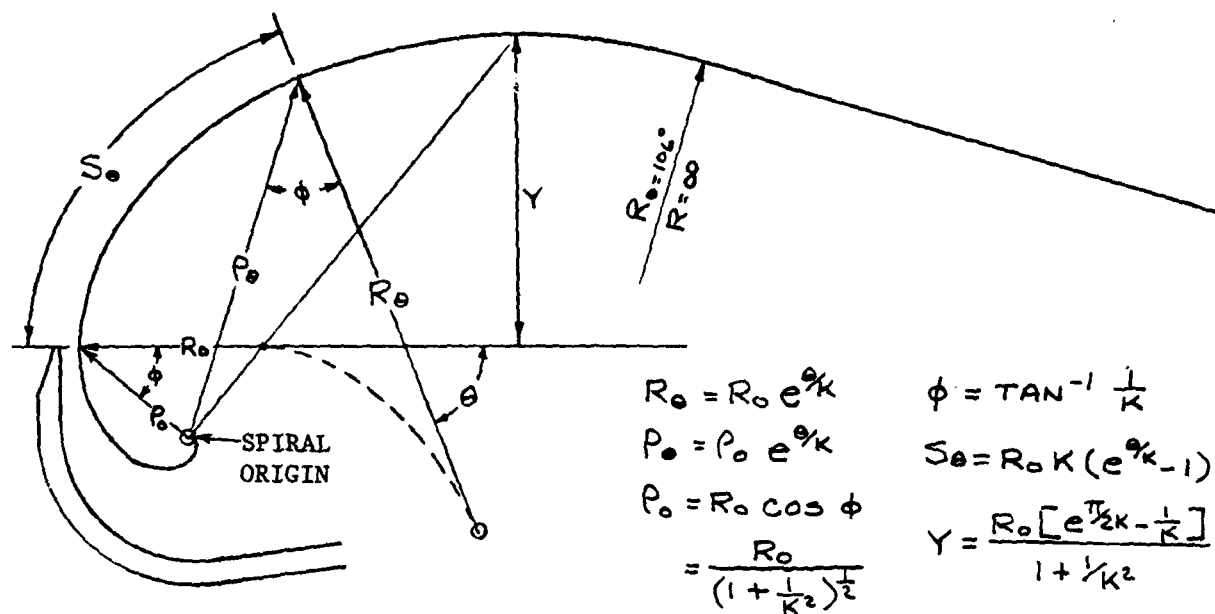
COANDA SURFACE PACKAGING

Figure 6-1.   Coanda Spiral-Definition & Packaging

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Table 6-1 Physical Parameters of Nozzles Tested

No.	K	R_0	t	R_0/t	$R_1 t$	R_{90}	Y	S_{90}/t
1	1.17	.518	.063	8.20	7.90	1.983	.890	27.1
2	1.20	.520	.065	8.00	5.40	1.925	.880	25.9
3	1.20	.520	.055	9.40	6.40	1.925	.880	30.7
4	1.30	.640	.073	8.79	6.87	2.143	1.037	26.7
5	1.50	.590	.063	9.37	7.94	1.681	.890	26.0
6	1.80	.440	.071	6.20	2.50	1.053	.618	15.5
7	1.80	.440	.071	6.20	7.00	1.053	.618	15.5
8	1.80	.682	.075	9.10	2.50	1.632	.958	22.8
9		.675	.077	8.80	3.50	.675	.675	13.8

Table 6-2 Spiral Shapes Tested

No.	K	R_0/t	R_1/t	W_{90}	W_{90}/R_{90}	C_V	$I = \dfrac{V_p}{V_0} \dfrac{W}{T}$
1	1.17	8.20	7.90	.46	.23	.81	4.18
2	1.20	8.00	5.40	.82	.43	.84	6.53
3	1.20	9.40	6.40	.53	.28	.85	4.77
4	1.30	8.79	6.87	.57	.27	.83	4.73
5	1.50	9.37	7.94	.38	.23	.76	4.00
6	1.80	6.20	2.50	.54	.51	.83	4.81
7	1.80	6.20	7.00	.43	.41	.88	3.71
8	1.80	9.10	2.50	.46	.28	.80	4.28
9	∞	8.80	3.50	.38	.56	.87	3.84

No. 3   Good performance – Higher than average C_V and I – Longest running length (S_{90}/t) – Safe W_{90}/R_{90}.

No. 4   Good C_V and I – largest $R_\theta = 90$ – fairly long running length – safe W_{90}/R_{90}.

No. 5   Lowest C_V, lower than Average I, lowest jet spreading.

No. 6   Good C_V, however, R_0/t small – W_{90}/R_{90} high – separates at NPR > 2.0:1.

No. 7   Good C_V, however, R_0/t small – W_{90}/R_{90} high – separates at NPR > 2.5:1.

No. 8   Lower than average C_V – separates at CTA < 60° at NPR > 2.5:1.

No. 9   Good C_V – low entrainment index – separates at CTA < 60° at NPR > 2.5:1.

Based on the above measurements and observations, a selection criteria was developed, which is discussed in detail in Reference (19) and summarized in paragraph 4.2 of this document, which allows a designer to select a log spiral compatible with packaging and nozzle pressure ratio requirements.

The Coanda shape tests discussed up to this point were all done at a model scale of approximately 0.2 to acquire as large a data base as possible in a short period of time. Configurations identified as potentially acceptable were then evaluated at full scale and the results of this testing is in the External Coanda Shape Evaluation portion of the Coanda segment model discussion (6.4.7).

6.3   INTERNAL RADIUS EVALUATION

In addition to the external Coanda shape testing discussed above, an isolated 20-inch span Coanda segment model was tested in an effort to identify the effect of variations in Coanda nozzle radius of approach and/or the nozzle lip convergence angle on Coanda component performance. Component lift and Coanda jet growth profiles were primarily utilized to quantify the performance levels of the configuration tested. The testing was conducted in two major phases. The objective of the initial phase was to isolate candidate configurations for values of radius of approach between $R_i/t = 2$ and $R_i/t = 5$. This range was selected based on past trade studies which indicate that acceptable values of R_i/t lie within this range from both a component performance and a design-packaging compatibility viewpoint. Additional information concerning the effect of nozzle lip convergence angle was also obtained during this phase of testing. All testing during Phase I was conducted in the Columbus Thermodynamics Lab. The second test phase was conducted at the Wright-Patterson Air Force Base Propulsion Labs. Higher values of compressor pressure and flow were available at this facility allowing a more adequate simulation of the pressure environment the component would be exposed to when installed in a thrust augmenter. The primary objective of Phase II was to evaluate the suitability of the internal radius ratio of 3.0 ($R_i/t = 3.0$) for aircraft hardware application.

6.3.1   Phase I

A great deal of isolated Coanda component testing had been conducted with an internal radius ratio of 4.0 ($R_i/t = 4.0$). While this radius of approach was found to be acceptable, design-packaging studies indicated slight improvements in hardware packaging were available through reductions in internal radius ratio. However, previous test experience indicated that radius ratios of approximately 2.0 ($R_i/t \approx 2.0$) could induce external Coanda jet flow separation. The purpose of this test was to identify the minimum internal radius which developed acceptable external Coanda flow quality while maximizing design-packaging benefits.

The model utilized in this test series was a 2-D isolated Coanda segment model with a 20-inch span. A K = 1.3 $R_o/t = 8.7$ external log spiral shape, and a nozzle exit height (t) of .3 inch (see Figures 6-2 and 6-3) were used for the evaluation of the internal radius ratios.
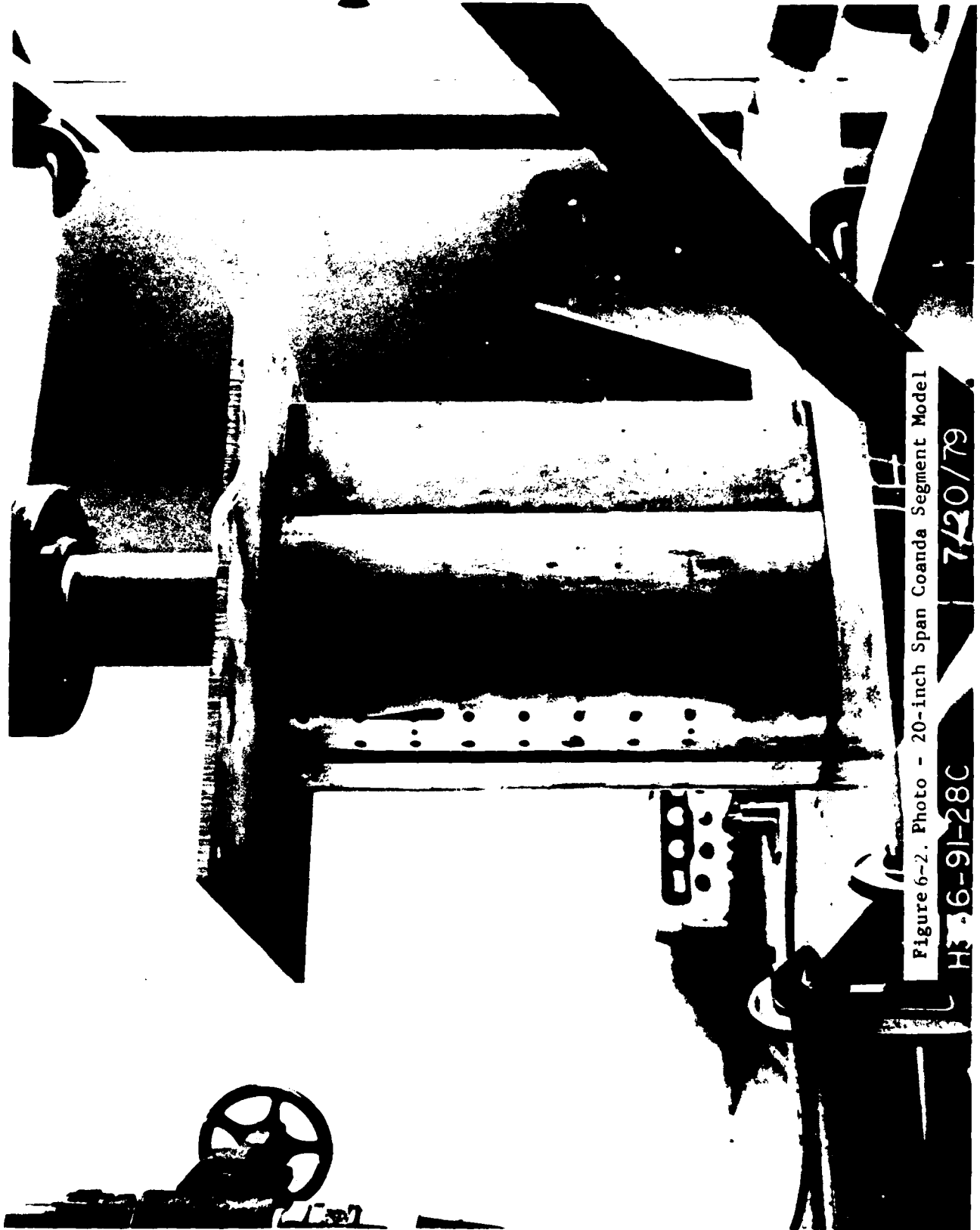
Figure 6-2. Photo - 20-inch Span Coanda Segment Model
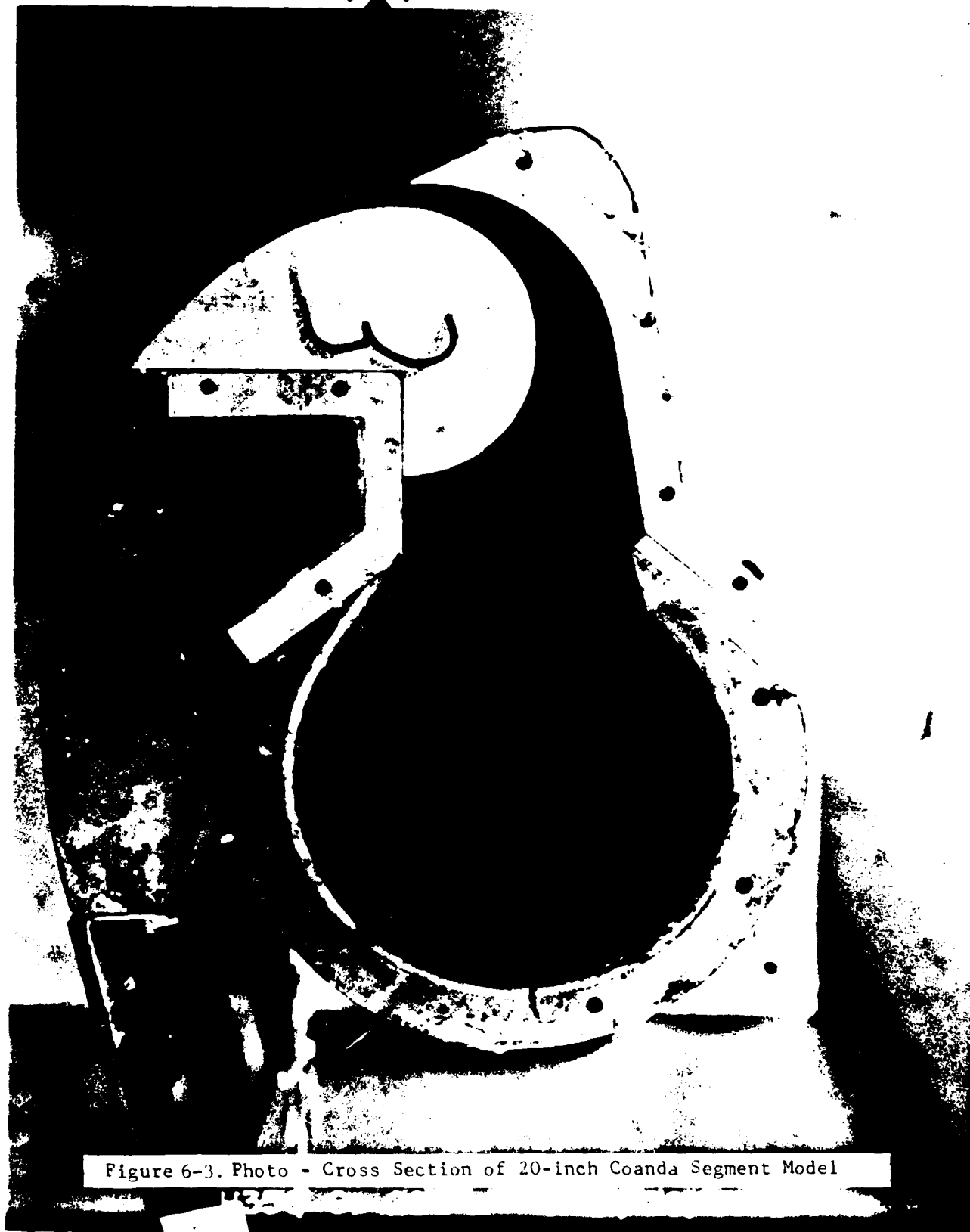
Figure 6-3. Photo - Cross Section of 20-inch Coanda Segment Model

During this phase of testing, three internal radius ratios were tested (R_1/t = 2, 3 and 5) with 0° nozzle lip convergence angle. To isolate the effect of increasing nozzle lip convergence angle, the R_1/t = 2.0 configuration was also tested with a 10° nozzle lip convergence. A sketch of the model with internal radius ratios from 2.0 to 5.0 installed is presented in Figure 6-4 along with the variation in upstream Mach number for each of the radius ratios pictured.

It should be noted that while an internal radius ratio of four is pictured in Figure 6-4, this radius ratio was not evaluated during this test series. A radius ratio of 4.0 was evaluated extensively during the Coanda segment model test program (see Reference (20)). The relative performance indicators utilized during this test series are listed below:

- Coanda jet growth total pressure profiles at nozzle exit, 3t, 5t, 7t, 9t, 15t, and 26t downstream of nozzle exit.

- Internal and external Coanda surface pressure taps located on the internal radius at 90°, 60°, 30° and .5 s/t upstream nozzle exit, and 5, 10, 19 and 26 s/t downstream of nozzle exit on the external Coanda shape.

- Nozzle flow coefficient (C_D) and component velocity coefficient (C_V) versus nozzle pressure ratio.

Comparisons of data of this type have typically been utilized to evaluate relative Coanda flow quality and performance during the previous model and full scale isolated Coanda component evaluation test programs and has been found satisfactory in isolating problems which may exist between individual components. Careful analysis of the data obtained during this test series resulted in the following conclusions (see Figures 6-5 to 6-7).

- No significant variations in turning efficiency (CTE) or relative jet growth can be inferred from the data presented in Figures 6-5 and 6-6. The small differences indicated in these comparisons are well within the data scatter for measurements of this type.

- Coanda surface pressure distributions indicate that none of the configurations tested exhibited any external Coanda flow separation up to the maximum nozzle pressure ratio tested, NPR $\leq$ 2.5. (see Figure 6-7).

Figures 6-8 to 6-10 present data comparisons developed for variations in the nozzle lip convergence angles of 0° and 10° of lip tilt for the R_1/t = 2.0 internal radius configuration. There is some indication that increasing the nozzle lip convergence angle may result in a slight reduction in external Coanda jet entrainment. This can be inferred from the small improvement in turning efficiency (CTE) shown in Figure 6-8 followed by a slight reduction in Coanda jet growth versus s/t presented in Figure 6-9. However, further study is necessary before any definite correlation between nozzle lip convergence angle and component entrainment can be developed. The small decrease in entrainment noted is more than

R$_i$/t Model

Blue Rig
R$_i$/t Model

EFFECT OF UPSTREAM NOZZLE
CONVERGENCE ON APPROACH
MACH NUMBER

R$_i$/t	Symbol
2.0	
3.0	
4.0	
5.0	

Convergence Mach No.

X Distance - Inches

Figure 6-4. Sketch of Cross Section Test Hardware and Effect of
Internal Radius on Nozzle Approach M$_N$

Figure 6-5. CTE vs s/t for R_i/t = 2, 3, 4, & 5.

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JET WIDTH VS DISTANCE

NPR = 2.0, K = 1.3, 0° LIP CONVERGENCE

□	$R_i/t = 2$	$R_0/t = 8.7$	R_i/t MODEL	
△	$R_i/t = 3$	$R_0/t = 8.7$		
○	$R_i/t = 5$	$R_0/t = 8.7$		
●	$R_i/t = 4$	$R_0/t = 11.3$	WPAFB MODEL	
◇	$R_i/t = 6.9$	$R_0/t = 8.7$	COANDA MODEL	

Figure 6-6. w/t vs s/t for $R_i/t = 2, 3, 4, 5$ and 6.9

DOWNSTREAM DISTANCE ~ (s/t)

JET WIDTH ~ w/t

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SURFACE STATIC PRESSURE VS SURFACE DISTANCE

NPR = 2.0:1, K = 1.3, R$_0$/t = 8.7, 0° L.P. CONVERGENCE

Figure 6-7. Surface Static Pressure Distributions for R$_i$/t = 2, 3 & 5

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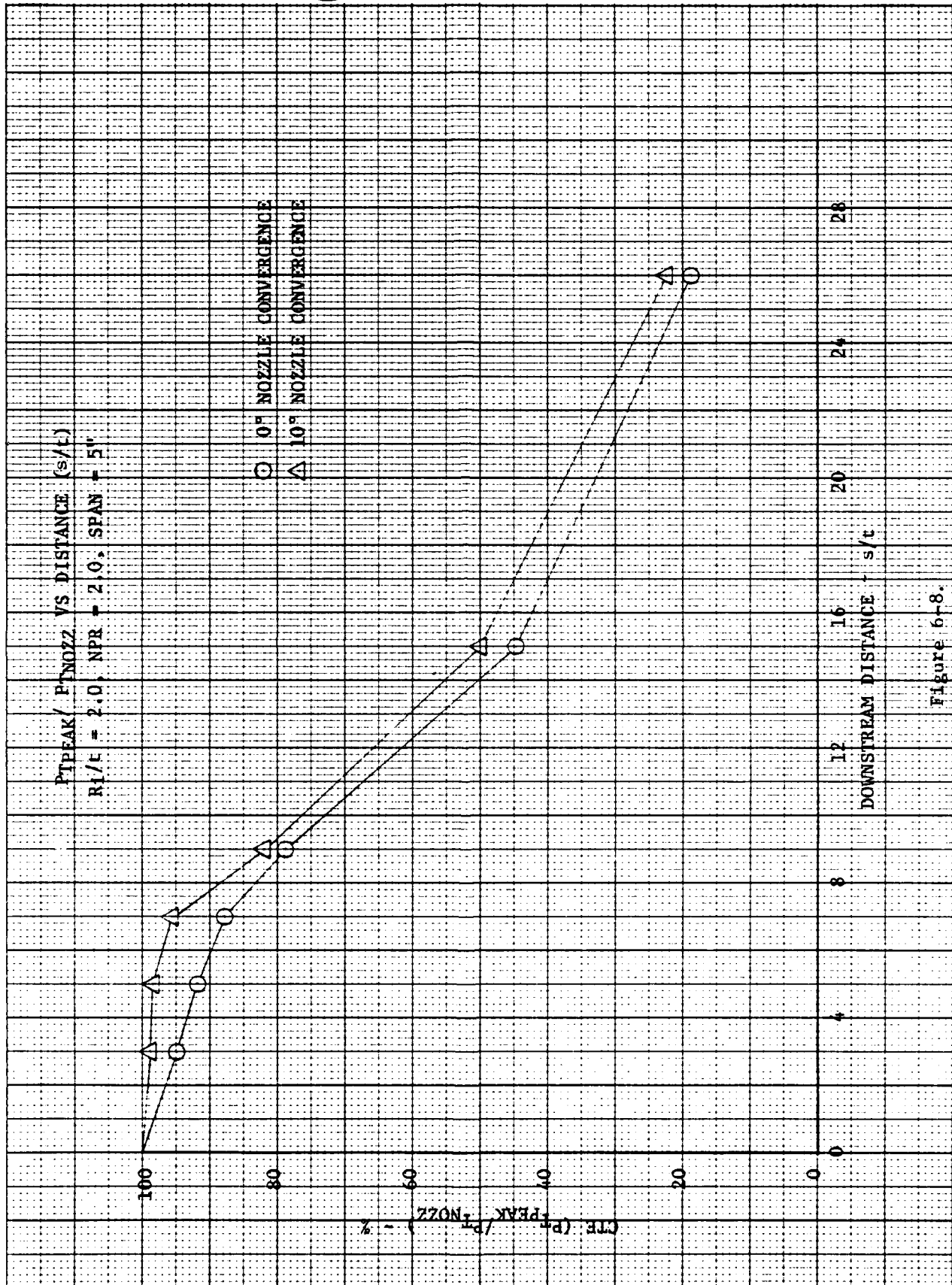
Figure 6-8.

Rockwell International
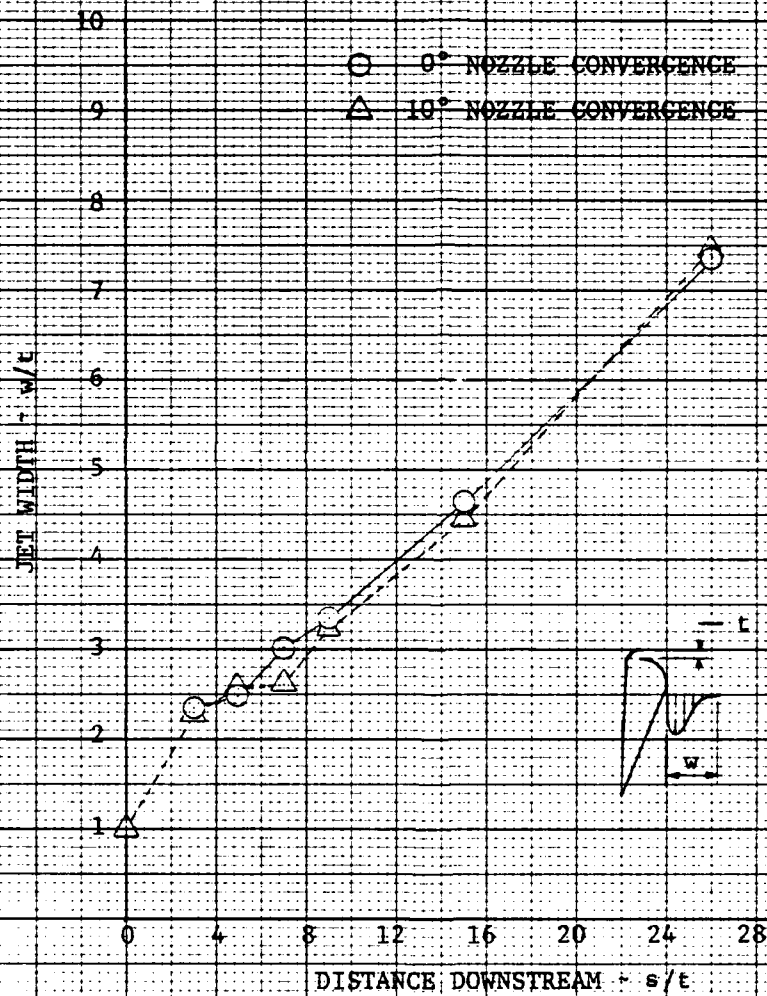
Figure 6-9 . w/t Vs s/t, $R_1/t = 2$, $0°$ & $10°$ Convergence

Rockwell International
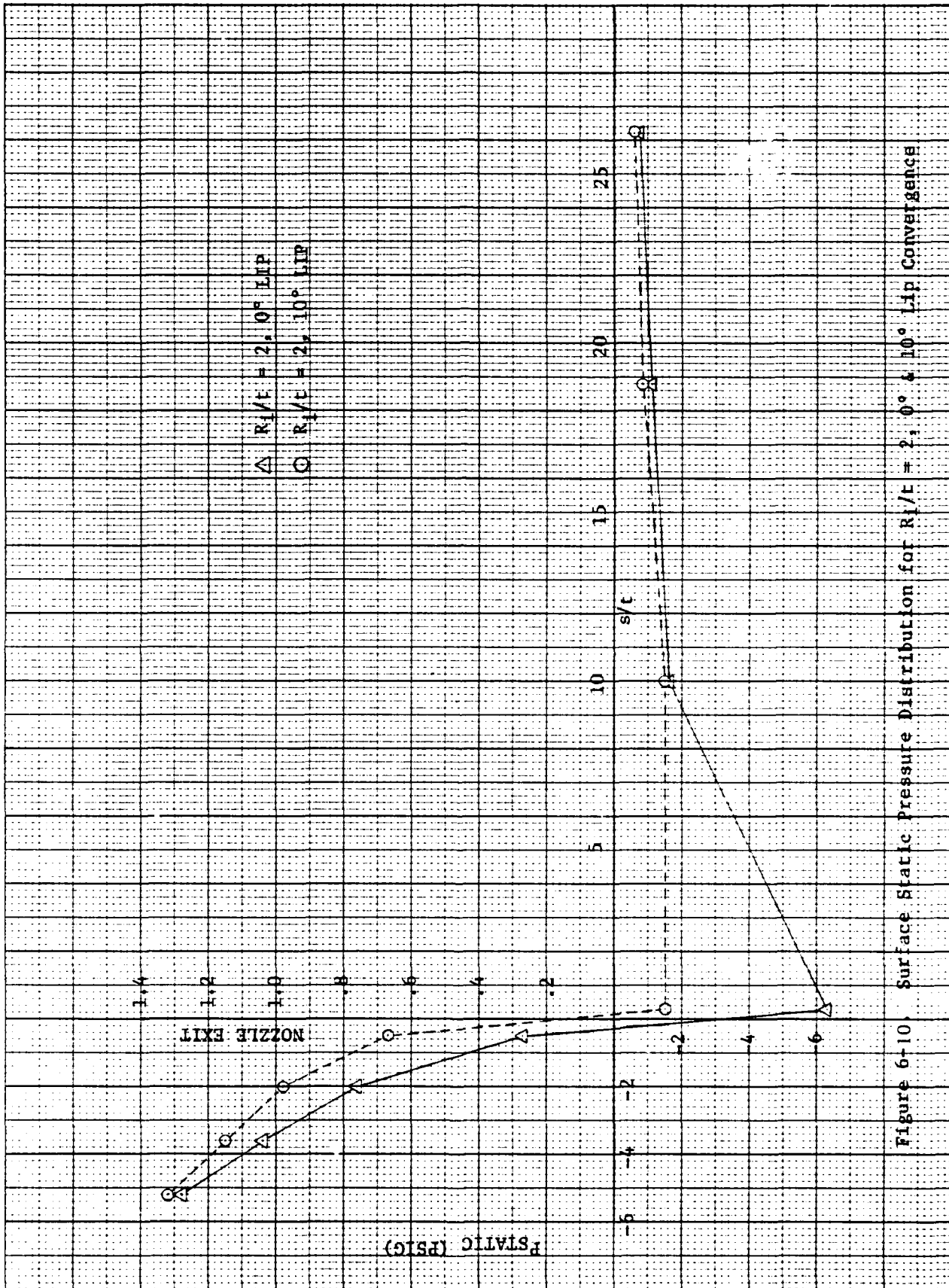
Figure 6-10. Surface Static Pressure Distribution for R_1/t = 2; 0°, & 10°; Lip Convergence

$\triangle$ $R_1/t = 2$, 0° LIP

$\bigcirc$ $R_1/t = 2$, 10° LIP

s/t

NOZZLE EXIT

P_{STATIC} (PSIG)

offset by the benefits obtained by slight degrees of nozzle lip convergence preventing the Coanda nozzles from distorting under pressure to a non-converging or converging-diverging (C-D) nozzle shape. The decrements in nozzle performance which may result from off-design C-D nozzles could be significantly larger than the performance decrease caused by slightly converging the nozzle as measured in this test. It is also possible that the slight decrease in entrainment noted may well be within test data measurement tolerances and be nothing more than data scatter.

In an effort to more fully simulate "in augmenter" flow conditions during isolated Coanda testing, it was necessary to test the isolated flap at nozzle pressure ratios somewhat higher than their typical "in augmenter" pressure levels. It was discovered that running isolated components at NPR's 25 to 30 percent greater than their normal operating level will approximate the pressure environment they will experience when installed in an augmenter shroud. Based on this information, additional tests were conducted in the internal radius variation test series at nozzle pressure ratios of approximately 3.0 (NPR $\approx$ 3.0). Phase I test data and concurrent design and packaging studies indicated that an internal radius ratio of 3.0 (R_i/t = 3.0) could provide the desired hardware packaging benefits while providing some margin of safety to assure acceptable flow quality and component performance. It should be noted that while the internal radius ratio of 2.0 (R_i/t = 2.0) appears satisfactory based on the data obtained during the initial test series, previous experience indicated that radius ratios in this range were responsible for unacceptable Coanda flow quality and performance (see References (17), (18), and (19)). Based on this data base the R_i/t = 2.0 was not utilized in future designs.

6.3.2 Phase II

In an effort to more fully evaluate an internal radius ratio of 3.0 (R_i/t = 3.0) and to further investigate the effect of nozzle lip convergence angle on component flow quality and performance, the R_i/t Coanda segment model utilized in the Phase I test series was installed in the isolated component test facility at Wright-Patterson Air Force Base (See Figures 6-11 and 6-12). A full description of this facility is presented in Section 10.0 of this document. It was necessary to conduct this phase of testing at the WPAFB facility in order to achieve model nozzle pressure ratios of approximately 3.0, providing a more realistic simulation of the flow conditions the components experience when installed in an augmenter.

Measurements performed and data obtained in order to quantify the models performance and flow quality included:

1. Model lift, nozzle mass flow, and nozzle pressure ratio (NPR) to be used to compute nozzle flow coefficient (C_D) and component velocity coefficient (C_V).

2. At four spanwise locations (2, 5, 7, 5 and 10 inches from inboard endwall) total pressure jet growth profiles were recorded at the nozzle exit plane and 3, 7, 9, 15 and 26 nozzle diameters (s/t's) downstream of nozzle exit.

Figure 6-11 Photo - Test Hardware Installed at WPAFB Facility

Figure 6-12 Photo - Test Hardware Installed at WPAFB Facility

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3. Coanda surface pressure taps were located midspan on both the
    internal and external Coanda surfaces as shown in Figure 6-4.
    This data was used to detect the presence of external Coanda flow
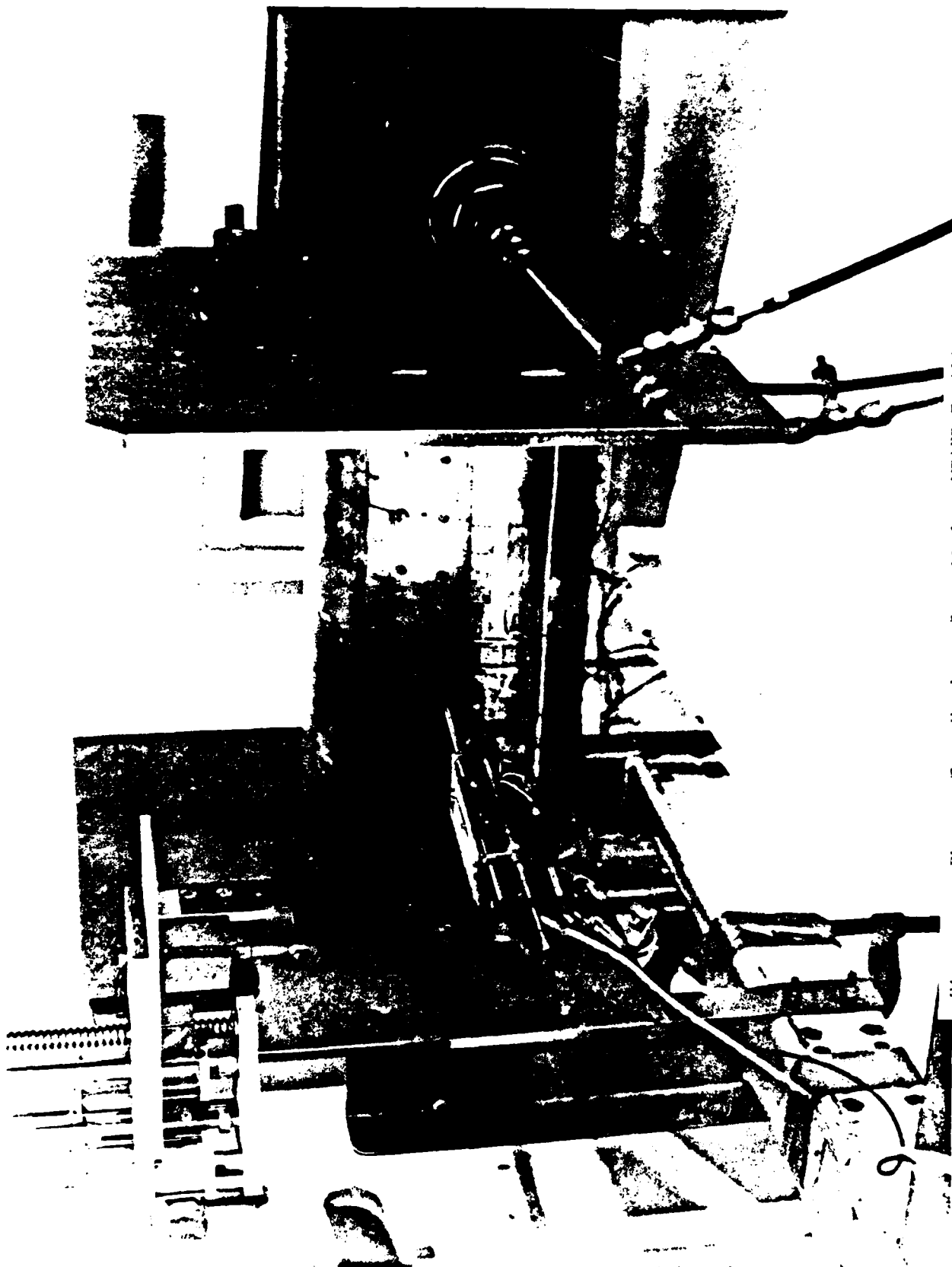   separations.

The initial configuration evaluated was the internal radius ratio of 3.0
(R_i/t = 3.0) with 5° of nozzle lip convergence. Isolated component
performance for this configuration is shown in Figure 6-13. Generally,
this configuration produced acceptable external flow quality and levels of
performance through the entire pressure range tested. There was no
indication of external Coanda flow separation and nominally the flow
quality appeared to be comparable to its R_i/t = 4.0 counterpart reported
in Reference (20). Unfortunately, direct comparisons of relative isolated
component performance are not possible for the R_i/t = 3.0 and the
R_i/t = 4.0 configurations. The R_i/t = 3.0 configuration was evaluated
on the 20-inch span Coanda section model described in this report while
the R_o/t = 4.0 investigation was conducted utilizing the 50-inch span
Coanda segment model described in Reference (20). Obviously, comparisons
of total component performance would be unrealistic due to the numerous
geometric differences in the respective test hardware.

Investigation of nozzle lip convergence angle is of interest because of
the tendency of the full scale aircraft nozzle to exhibit varying degrees
of nozzle lip expansion proportional to NPR and gas temperature. This
expansion can result in distortions in the Coanda nozzle shape which are
severe enough to form non-converging or C-D type nozzles. Since these
expansions are random and proportional to NPR and gas temperature, it is
unlikely that the NPR (Mach number) and degree of C-D would match. In
fact, when the character of C-D nozzles are not matched to their NPR,
their performance will be off design and a simple convergent nozzle will
often exhibit superior levels of performance. Additionally, the potential
gains in nozzle performance possible through the use of C-D nozzles are
quite small at NPR ≈ 3.0. Finally, the degree of complexity and tolerances
required to match a C-D nozzles characteristics is beyond the practical
scope of our current hardware fabrication techniques. Matching the degree
of expansion to the nozzle C-D versus NPR charactersitics would be
extremely difficult. Therefore, it would appear that it is quite
desirable to prevent the nozzle from expanding during test since current
materials and fabrication methods do not allow complete elimination of
nozzle expansion. Some method must be identified which will eliminate the
possibility of forming a C-D channel when the nozzle expands.

In order to prevent the nozzle from distorting to some non-optimum shape,
a small degree of additional nozzle lip convergence (lip tilt) can be
built into the nozzle exit appproach geometry. The lip tilt will be
reduced during operation - the amount of reduction being proportional to
operating NPR and gas temperature. Theoreticaly it is possible to match
the degree of nozzle lip expansion and nozzle lip tilt so that at the
typical normal operating pressure ratio the nozzle would distort to the
optimum nozzle shape and area. Unfortunately, the hardware tolerances
required to accomplish this option are not possible with current
fabrication techniques. It is also difficult to define the aircraft's

46 1320

VARIATION OF C_V & C_D WITH NPR

TEST CONFIGURATION R_i/t = 3.0, 5° LIP CONVERGENCE

1.0

.9

C_V

.8

$$C_V = \frac{\text{LOAD STAND LIFT}}{\text{NOZZLE ISEN. THRUST}}$$

.7

1.0　　1.5　　2.0　　2.5　　3.0

NPR

TEST	SYM
129	○
130	□
131	△

1.0

.9

C_D

.8

$$C_D = \frac{\text{VENTURI MASS FLOW}}{\text{IDEAL NOZZLE MASS FLOW}}$$

1.0　　1.5　　2.0　　2.5　　3.0

NPR

Figure 6-13　Component C_V & Nozzle C_D Vs NPR R_i/t = 3, 5° Lip Convergence Angle

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description of the WPAFB test facility is presented in the facilities
section of this report. These tests were conducted in three basic phases
(see Figure 6-14) and included variations in the following variables.

Flow Variations

  - Nozzle Pressure Ratio
  - Internal Feed Duct Mach Number

Model Variations

  - Feed Duct Location Relative to Coanda Nozzle Exit Plane
  - External Coanda Shapes
  - Internal Coanda Shapes
  - Internal Flow Turning Devices

The effects of variations in the variables on total component performance
were isolated through direct measurements of model loads, nozzle airflow,
and pressure characteristics. A traversing pressure rake was employed for
measurements of pressure profiles at various s/t's along the Coanda
surface. Various flow visualization techniques were employed to determine
component flow quality and isolate the degree of spanwise flow angularity.

6.4.3   Model

The model tested shown in Figure 6-15 simulates the XFV-12A lefthand wing
elevon inboard section (initial 50 inches of span) and is completely
defined in Reference (20). In order to simulate aircraft feed conditions
on the model, an S-shaped feed duct similar to its aircraft counterpart
was fabricated and attached directly to the model mounting plate,
Figure 6-15. The basic model design allowed for a high degree of
flexibility providing for variations in feed duct location relative to
nozzle exit plane, as shown in the sketch on Figure 6-16. A manually
adjusted dump door, Figure 6-17, was located at the outboard end of the
model to provide variations in inlet Mach number.

Three different Coanda shapes were installed during the test program. A
sketch presenting geometric comparisons of the three shapes is presented
in Figure 6-18. Following pressurization of each Coanda shape, plaster
castings at 8, 24, and 42 inches from the inboard endwall were made to
verify that the shape contours were within specified design tolerances.

Direct hardware modifications were accomplished to evaluate internal
radius ratios of 3.0 and 4.0.

6.4.4   Baseline Performance

Prior to any attempt to optimize the overall performance of the XFV-12A's
elevon utilizing the Coanda segment model, it was imperative that the
Coanda segment model in its baseline configuration simulate the performance
and flow characteristics of the existing wing elevon. The primary purpose

Figure 6-14 . Coanda Segment Model Test Summary

	APRIL	MAY	JUNE	JULY	AUGUST	SEPTEMBER

WPAFB Facility Activation and Model Preparation — 24 4

Initial Facility Checkout Runs — 7 17

Phase I
- Basic Aircraft Configuration — 18 23 Runs 1-12
- Feed Position 1" Below — 31 7 Runs 13-23
- Feed Position 2" Below — 8 12 Runs 24-35
- Feed Position 4" Below — 14 15 Runs 36-42

Phase II
- Install K = 1.3 Shape @ Columbus — Runs 43-49
- Evaluation Runs at WPAFB — 5 23 Runs 50-78
- Install K = 1.17 Shape @ Columbus — 7 20
- Evaluation Runs at WPAFB — Runs 79-102

Phase III
- Install R_1/t = 3.0 Internal Radius — 7 17
- Evaluation Runs at WPAFB — Runs 103-116

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FIGURE 6-15   FULL SCALE ELEVON COANDA SEGMENT MODEL

WPAFB COANDA SEGMENT MODEL

NOZZLE

COANDA SURFACE

FEED DUCT

(ALTERNATE FEED POSITIONS SHOWN)
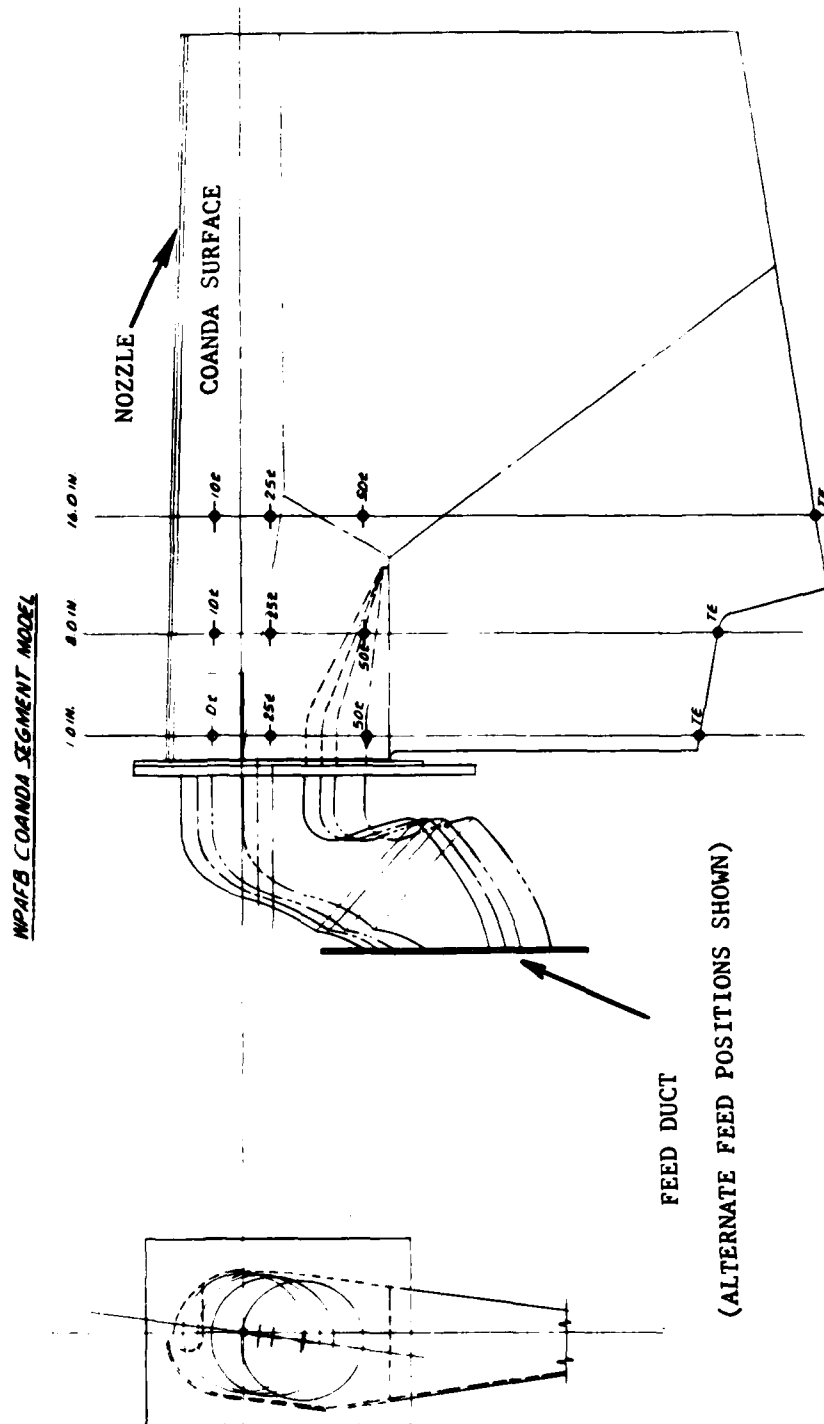
FIGURE 6-16 FEED DUCT POSITIONS

NOZZLE BULB SEAL

OUTBOARD END PLATE

INBD ENDPLATE

METAL STRAPS

ADJUSTABLE DUMP DOOR

FIGURE 6-17. DUMP DOOR

ELEVON INLET FEED DUCT

K = 1.3 SPIRAL

PROTO 1

K = 1.17 SPIRAL
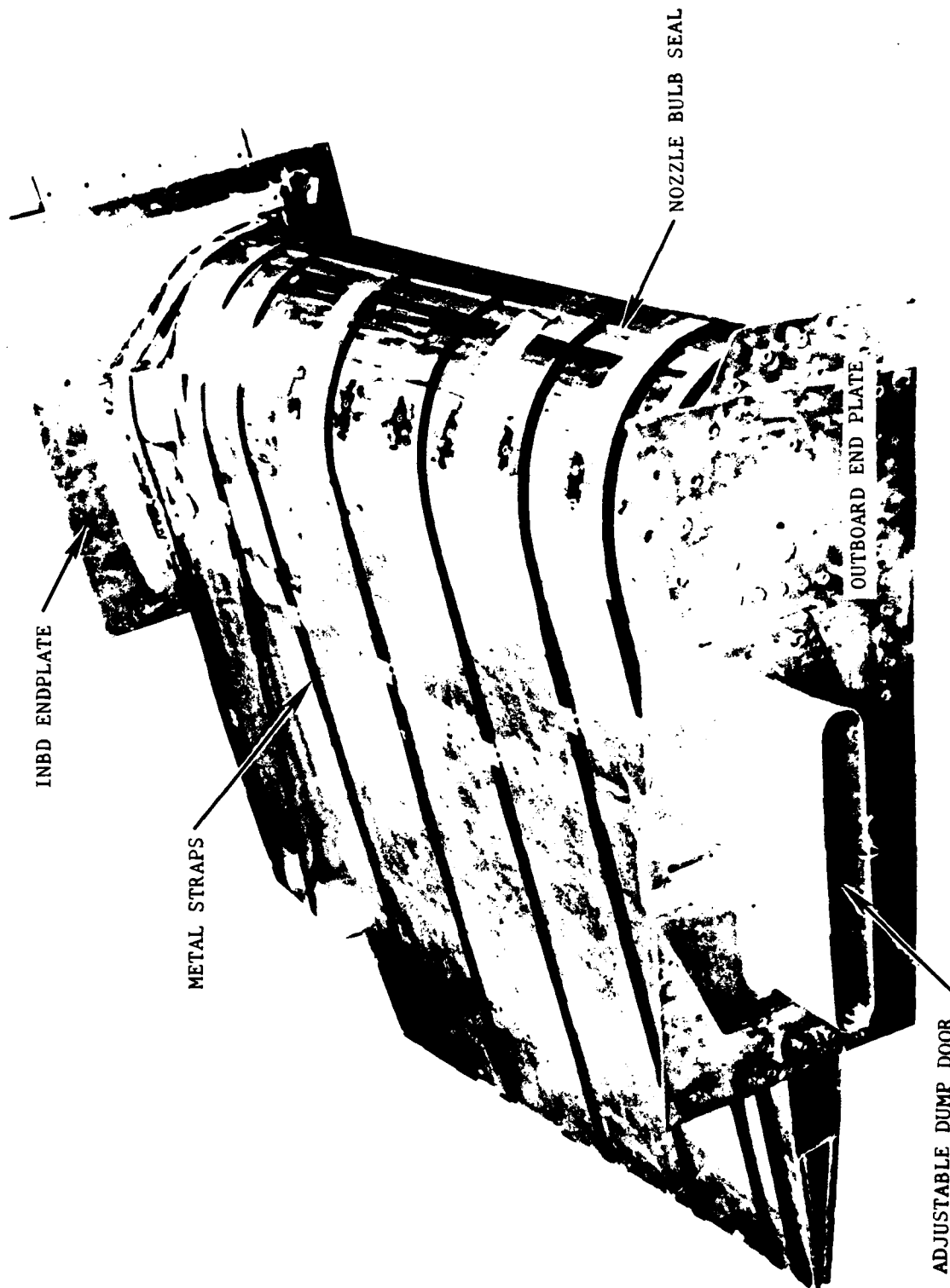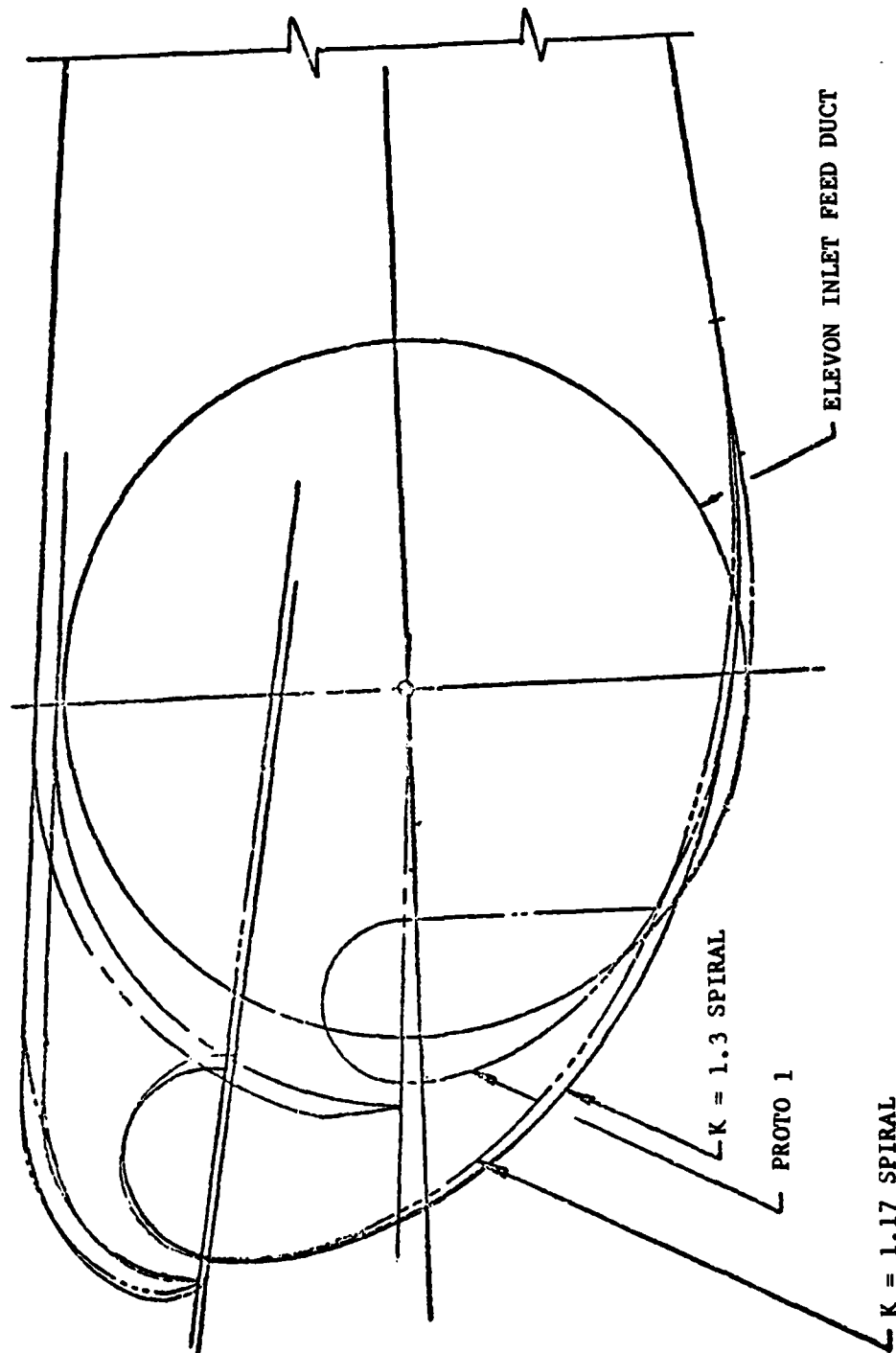
FIGURE 6-18 . COANDA SHAPE COMPARISON

of the initial test series was to establish the performance
characteristics of the baseline configuration to assure simulation of the
Proto 1 elevon. Comparison of this data, Reference (20), indicates the
performance similarity.

As presented in Figure 6-19, the spanwise nozzle total pressure
distribution for the Coanda segment model and the XFV-12A elevon are very
similar in nature. Both exhibit high pressure peaks in the inboard
corner, followed by a flow defect outboard of the interface turning vane,
and finally flat profiles from approximately ten inches outboard.

Flow visualization studies conducted on both components indicated that
both exhibited similar flow characteristics; i.e., a high degree of
spanwise flow angularity in the inboard region of each component.
Figures 6-20 and 6-21 present flow visualization photographs which are
representative of these tests.

Based on the foregoing information as well as analysis of pressure profile
data, it can be concluded that the segment model successfully represented
the XFV-12A aft flap.

6.4.5   Location of Feed Duct

Variations in the location of the feed duct relative to the Coanda nozzle
exit plane were evaluated for sensitivities of Coanda flow quality and
performance. Basic variations of one, two, and four inches were conducted
as shown in the sketch, Figure 6-16. Test data indicated that relocating
the Coanda feed duct one and two inches below the Coanda nozzle exit plane
significantly improves component flow quality. At the two inch position,
however, a notable drop in the nozzle exit total pressure was observed
from the inboard endwall to the outboard end of the interface turning vane
as shown in Figure 6-22. Inspection of the model hardware indicated that
when the feed duct was located at the two inch position, the interface
turning vane was actually blocking flow from the inboard region of the
Coanda nozzle exit. As shown in Figure 6-23, removing the turning vane
vastly improved the pressure distribution in this region. While the
interface turning vane proved useful when the feed duct was located in the
baseline position, it is not required and is in fact harmful when the feed
duct is located in the two inch position. Moving the feed duct to the
four inch position failed to produce any further notable improvement in
Coanda flow quality. Variations in C_V and C_D as a function of feed
duct location are presented in Figure 6-24. Additional confirmation of
the improved flow quality with the relocated feed duct (without turning
vane) was the jet pressure profile development along the surface,
Figure 6-25.

6.4.6   Effect of Inlet Mach Number

Opening the dump door at the outboard end of the model produced a
variation in feed duct mass flow and Mach number at constant nozzle
pressure ratios. The objective of this procedure was to vary Mach number
to simulate variations in feed duct area. As shown in Figure 6-26,
reducing the feed duct Mach number from M = .235 representative of the

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NOZZLE TOTAL PRESSURE DISTRIBUTION COMPARISON

WPAFB - RUN 19 - BASELINE CONFIGURATION

THERMO LAB - RUN 11 - AIRCRAFT SIMULATION

WPAFB - RUN 19

THERMO LAB - RUN 11

RELATIVE PRESSURE (% x 100)

1.0

0.5

0

SPAN (INCHES)

0   4   8   12   16   20   24

TURNING VANE
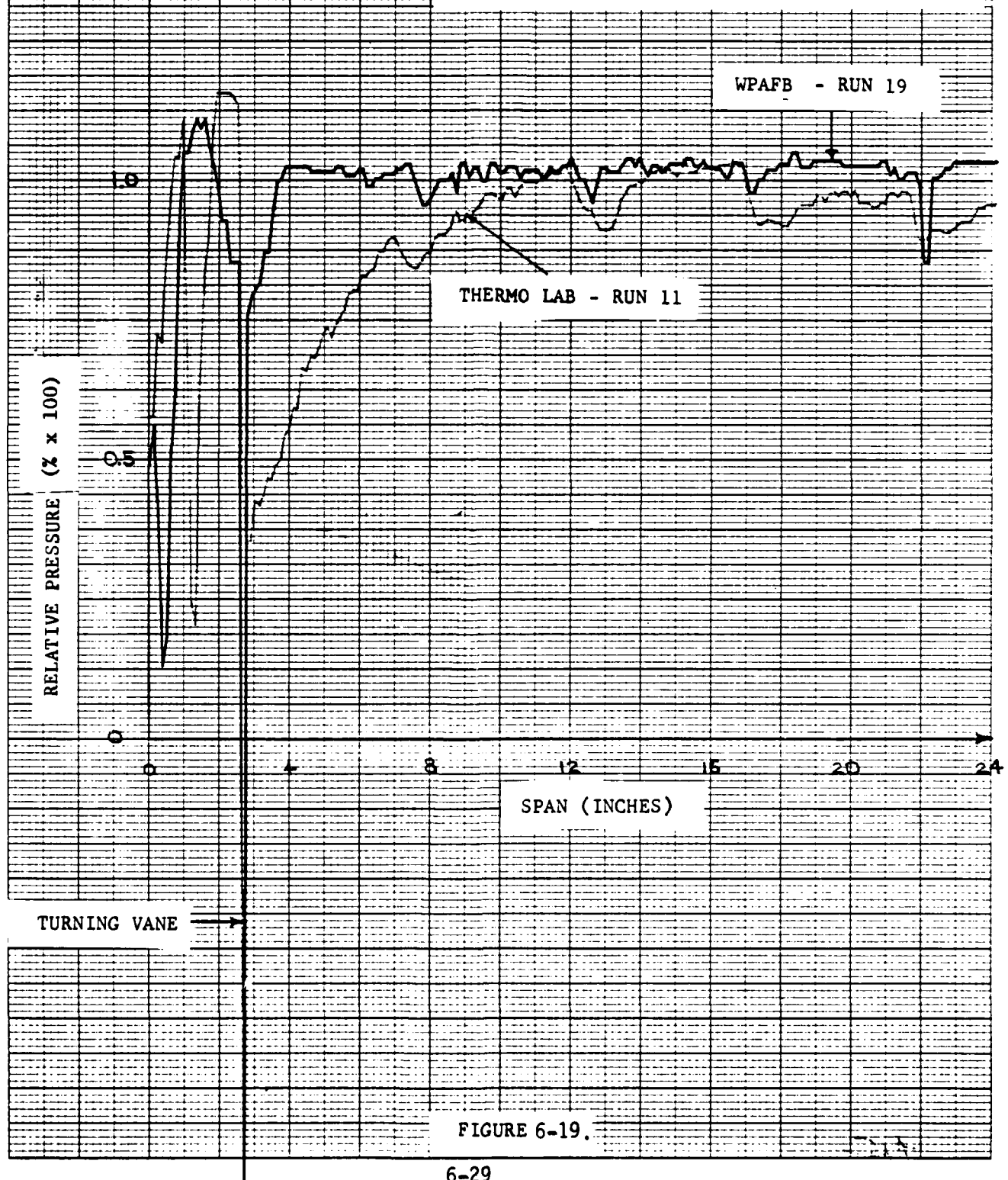
FIGURE 6-19.

(A)　INBOARD CORNER

(B)　6 IN. OUTBOARD
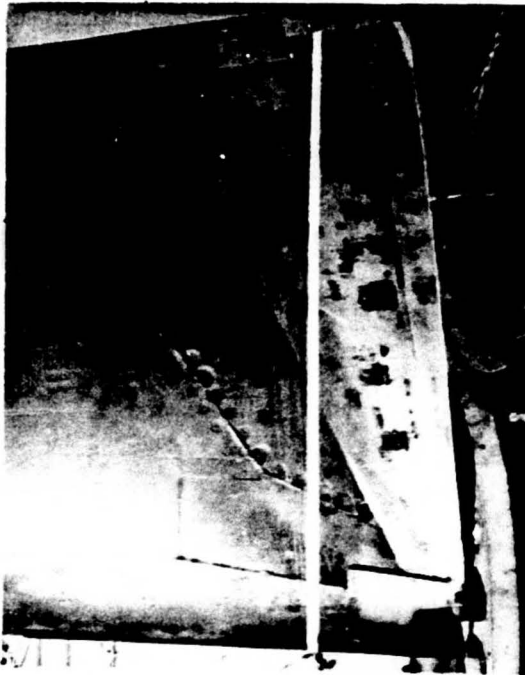
(C)　8 IN. OUTBOARD

(D)　14 IN. OUTBOARD

FIGURE 6-20. FLOW VISUALIZATION. ELEVON
IN THERMO LAB.

FIGURE 6-21 FLOW VISUALIZATION. COANDA
SEGMENT MODEL. RUN 34

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NOZZLE TOTAL PRESSURE DISTRIBUTION VS. SPAN

RUN 19 - BASELINE CONFIGURATION
RUN 32 - FEED DUCT 1 INCH BELOW COANDA PLANE
RUN 37 - FEED DUCT 2 INCHES BELOW COANDA PLANE
RUN 46 - FEED DUCT 4 INCHES BELOW COANDA PLANE
NOMINAL PRESSURE RATIO = 2.05
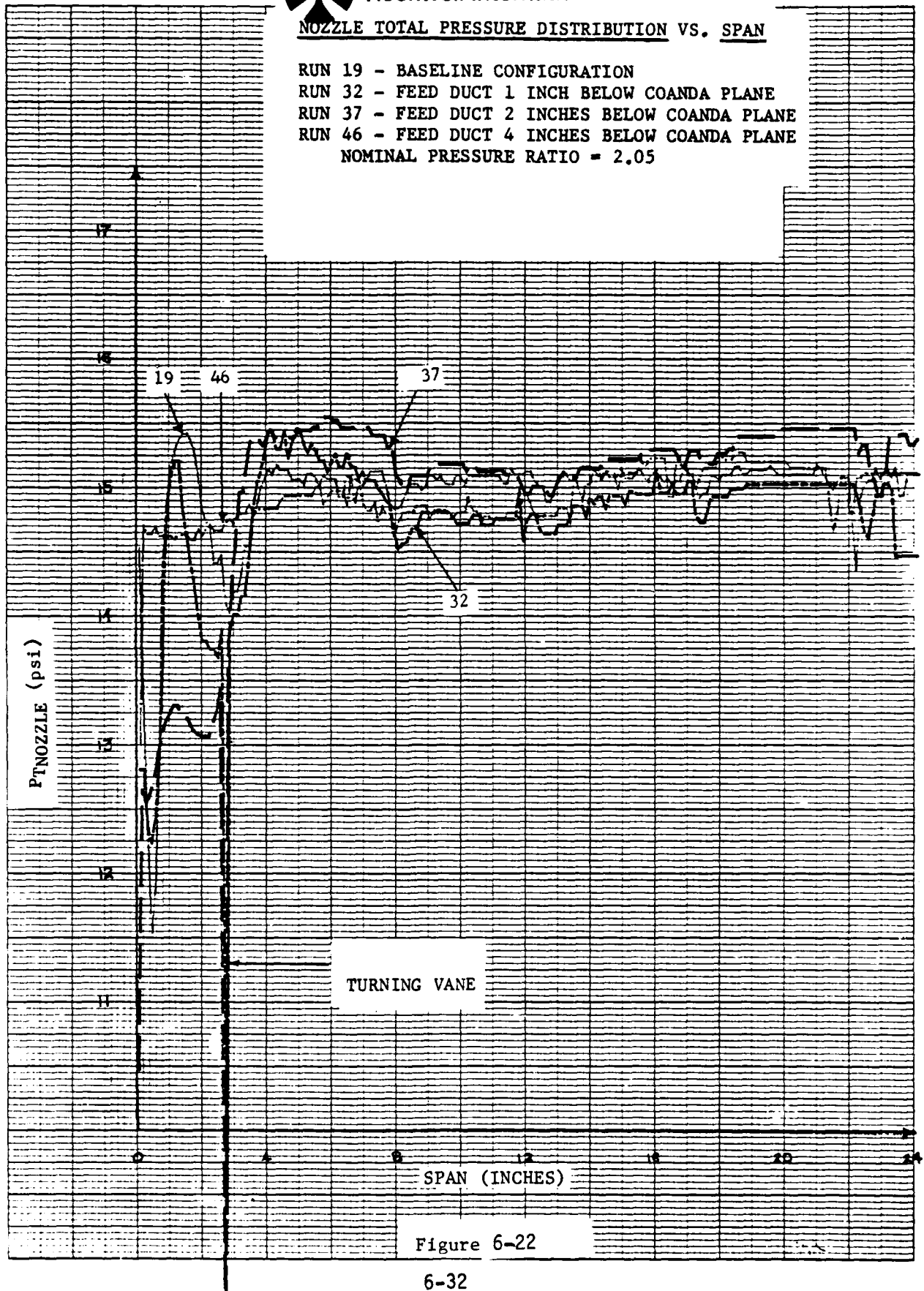
Figure 6-22

NOZZLE TOTAL PRESSURE DISTRIBUTION VS. SPAN

RUN 37 - TURNING VANE PRESENT
RUN 41 - TURNING VANE REMOVED
NOMINAL PRESSURE RATIO = 2.05
DUMP DOOR CLOSED

FEED DUCT 2 INCHES BELOW COANDA PLANE

Figure 6-23

FLOW AND VELOCITY COEFFICIENTS

⊙ RUN 19 - BASELINE CONFIGURATION
⊙ RUN 32 - FEED DUCT 1 INCH BELOW COANDA PLANE
△ RUN 37 - FEED DUCT 2 INCHES BELOW COANDA PLANE
⊡ RUN 46 - FEED DUCT 4 INCHES BELOW COANDA PLANE

CD

NOZZLE PRESSURE RATIO

CV

NOZZLE PRESSURE RATIO
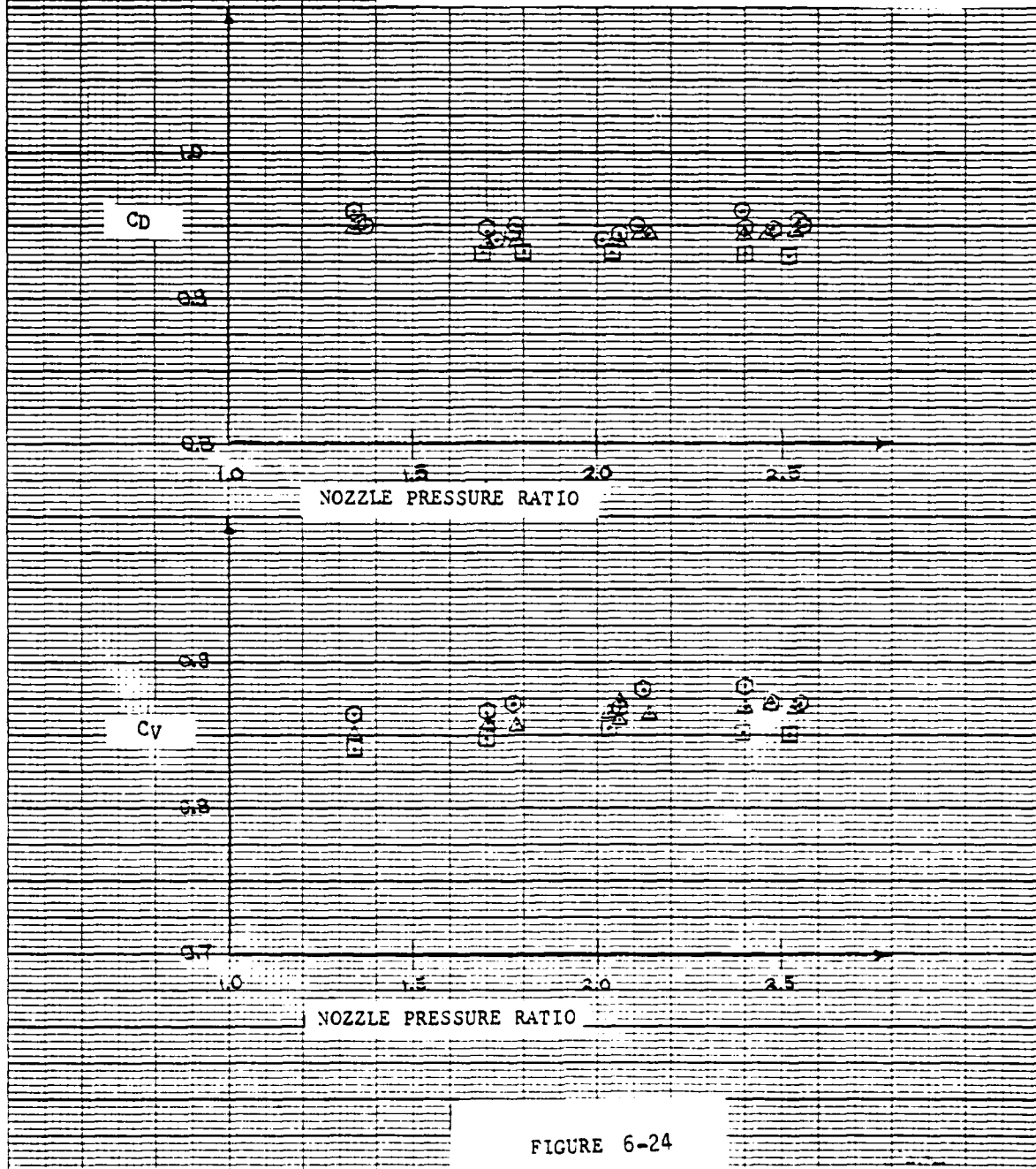
FIGURE 6-24

A COMPARISON PLOT OF THE FOUR DIFFERENT
FEED DUCT CONFIGURATIONS AT THE 16 INCH
SPAN STATION

⊙ RUN 23 - BASELINE CONFIGURATION
◐ RUN 35 - FEED DUCT 1 INCH BELOW COANDA PLANE
▲ RUN 39 - FEED DUCT 2 INCHES BELOW COANDA PLANE
● RUN 42 - FEED DUCT 2 INCHES BELOW COANDA PLANE
▣ RUN 48 - FEED DUCT 4 INCHES BELOW COANDA PLANE
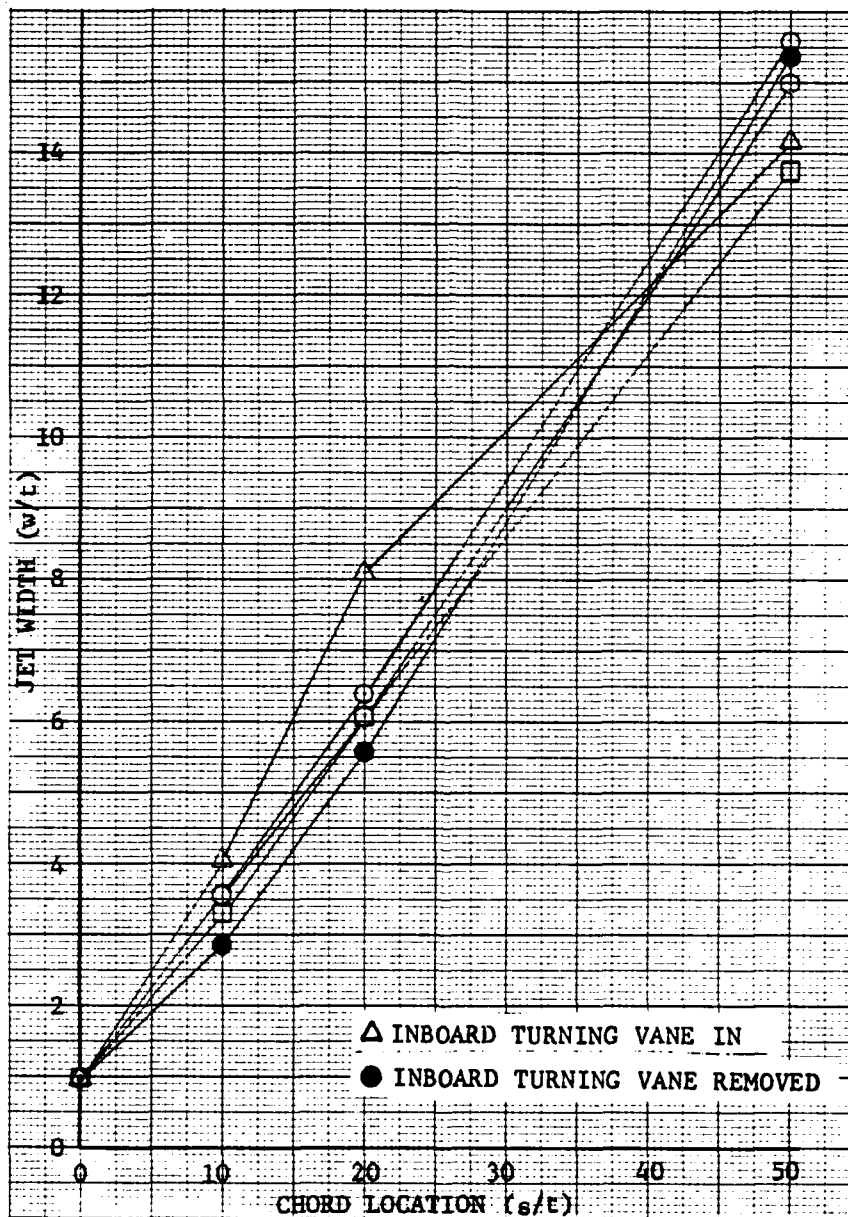
Figure 6-25   Jet Width Vs Chord 16" Span Station

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NOZZLE TOTAL PRESSURE DISTRIBUTION VS. SPAN

A COMPARISON OF INLET MACH NUMBERS

RUN 19 - INLET M_N = 0.135
RUN 22 - INLET M_N = 0.235

NOMINAL PRESSURE RATIO = 2.05

SPAN (inches)
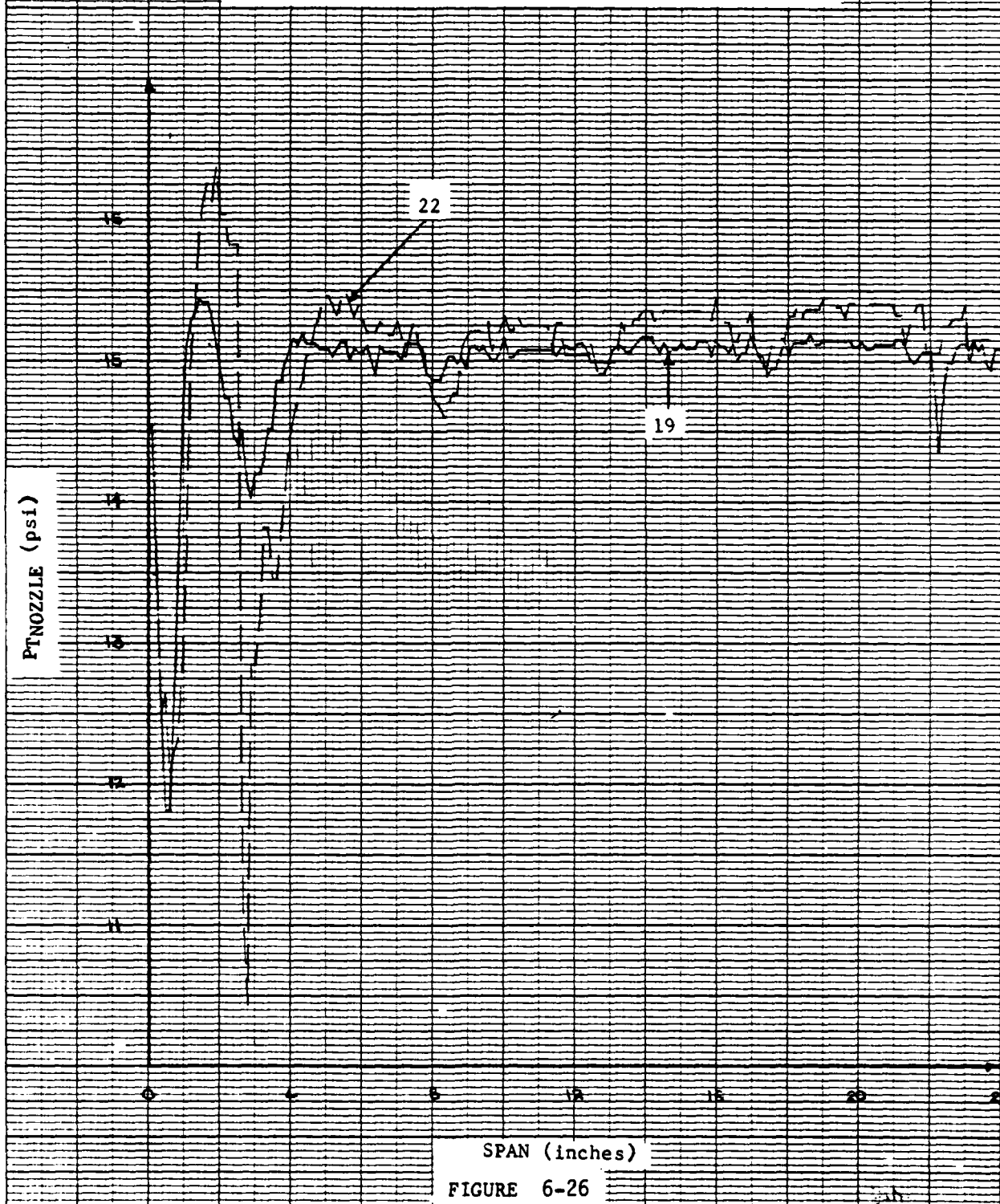
FIGURE 6-26

XFV-12A elevon to M = .135 failed to produce any notable improvement in spanwise nozzle total pressure distribution. In fact, no improvement in any of the Coanda evaluation parameters was noted during the Mach number variation test series. Based on this information, it was concluded that no improvement in flow quality could be expected by simply increasing the feed duct area, i.e., reducing feed duct Mach number.

6.4.7   External Coanda Shape Evaluation

Phase II testing of the segment model was conducted to evaluate full scale performance of Coanda shapes established by model scale tests as having potentially improved performance. Also, an attempt was made to improve the flow quality at the inboard end which was found to have a large spanwise component on Proto 1. Reference (18) documents previous tests of these Coanda shapes on a scale model, and some data are included here for comparison purposes.

The models tested were full scale tapered sections of the inboard 50 inches of the L/H wing elevon. The gap settings were chosen to give about the same R_o/t (10.5 - 11.5). The R_i/t was set at a nominal value of four. Comparison of the Coanda shapes were conducted by utilizing the following four performance criteria:

    1. Flow and velocity coefficients
    2. Jet growth
    3. Turning efficiency
    4. Nozzle pressure distribution

Figure 6-27 is a comparison plot of the flow (C_D) and velocity (C_V) coefficients obtained for the three Coanda shapes. Little differences in flow coefficients were noted for the three Coandas which is to be expected since flow coefficients are primarily dependent on internal nozzle design.

The velocity coefficient plot indicated about two percent better performance of the K = 1.3 spiral over the K = 1.17 spiral and a similar increment between the Proto 1 and K = 1.3 spiral. The velocity coefficients for the K = 1.7 and K = 1.3 spirals compare favorably with the values obtained on scale models reported in Reference (18). Isolated points are plotted on Figure 6-27 as solid symbols.

The jet growth, an indication of entrainment, was measured for the three Coanda shapes and the results are compared in Figure 6-28. This comparison was conducted at span station 16 to avoid the jet disturbing effects of the auxiliary corner blower. Little difference was noted in the jet growth characteristics for the three Coanda shapes. Very good correlation with scale model data was obtained. Scale model test data (Reference (18)) is plotted in Figure 6-28 as solid symbols.

The Coanda Turning Efficiency (CTE) as defined (PT_{peak}/PT_{nozzle}) is a measure of the pressure remaining in the jet after being turned by the Coanda some given distance. A comparison plot of the Proto 1 simulation

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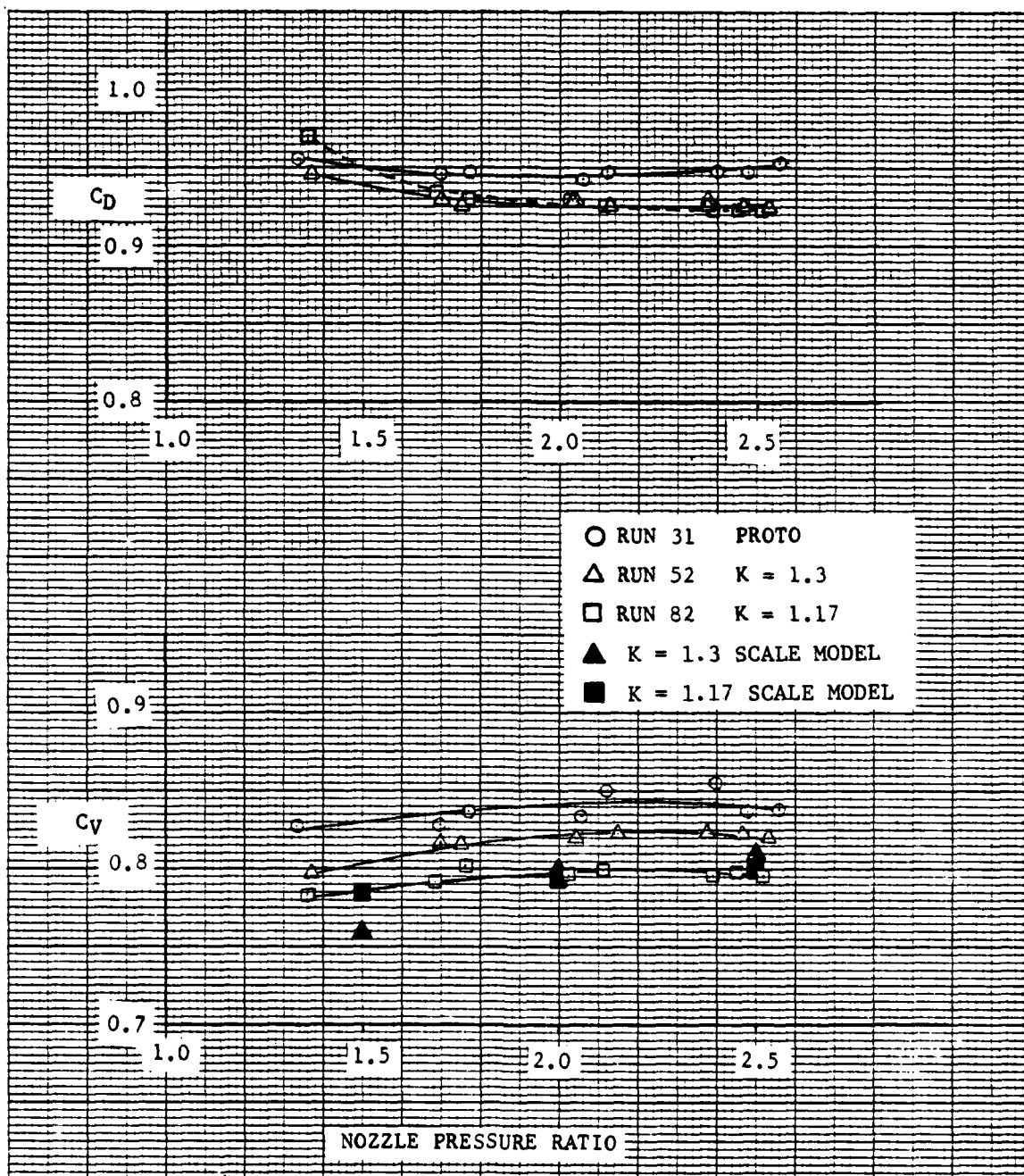
Figure 6-27 Flow and Velocity Coefficients Comparison
WPAFB Coanda Segment Model

NR8₁H-50

JET GROWTH VS. CHORD DISTANCE COMPARISON
SPAN STA 16

FIGURE 6-28

6-39
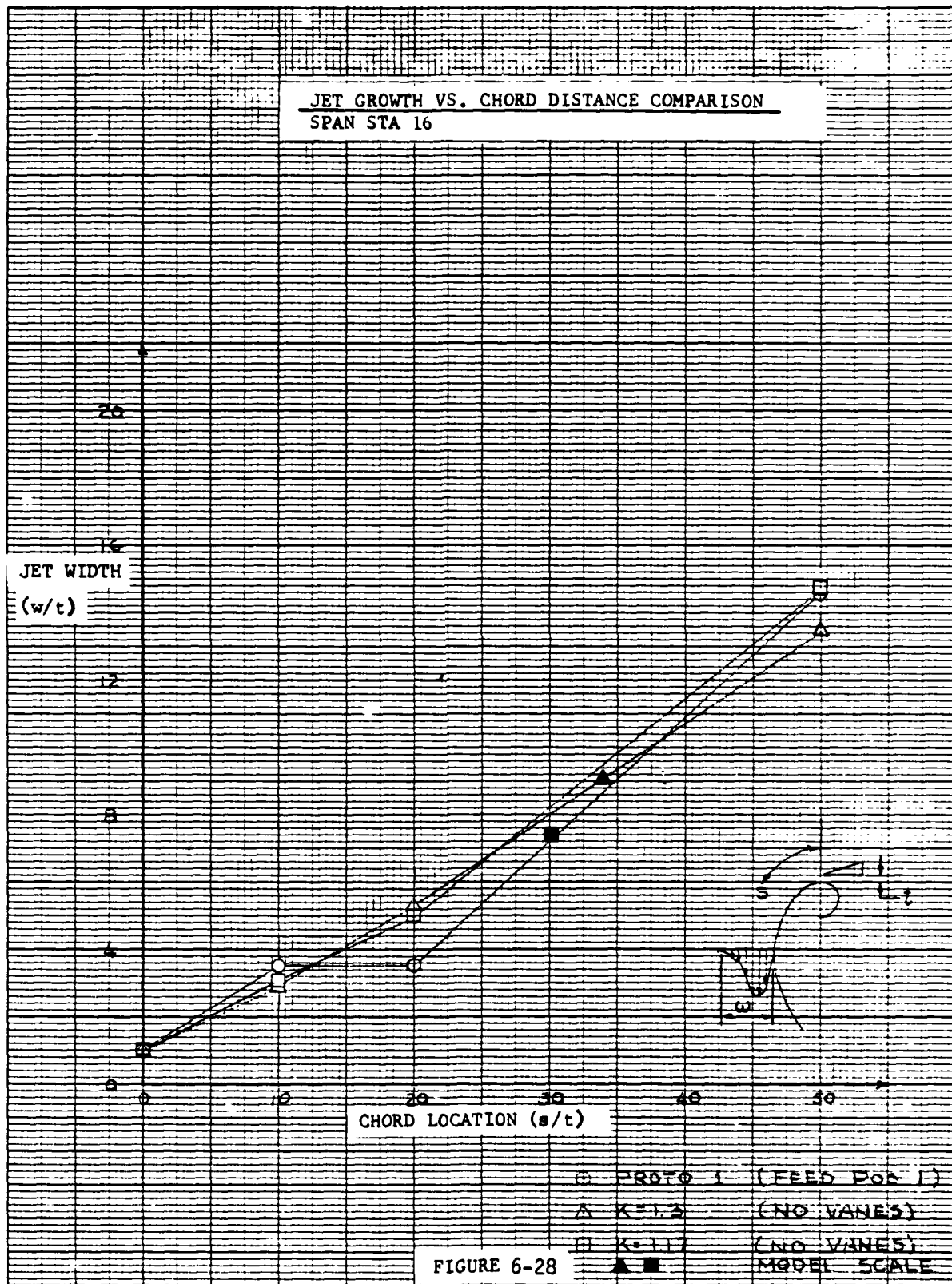
and the K = 1.17 Coanda with three vanes in the dedicated nozzle is
presented in Figure 6-29. At ten nozzle gaps (10t) from the nozzle the
K = 1.17 Coanda is clearly better at all span stations and at 50t from the
nozzle the benefits of the vanes in the auxiliary corner blower are very
much evident for the first five inches of span. Figure 6-30 is a similar
plot comparing the Proto 1 simulation to the K = 1.3 Coanda with a single
turning vane in the auxiliary blower. Similar results were observed;
i.e., at 10t the K = 1.3 spiral was better at all span stations and at 50t
very much better CTE of the inboard five inches.

Attempts were also made during this phase of testing to improve the nozzle
pressure distribution, especially at the inboard end. Figure 6-31 shows
improvements made to the main nozzle pressure distribution by the addition
of vanes in the auxiliary corner blower (examples of corner blower vane
installation may be seen in Figures 6-32, 6-33, and 6-34). This primary
nozzle pressure distribution improvement is probably related to creation
of a more favorable airflow at the inlet duct by the vane installation in
the corner blowers. Improvements to the primary nozzle pressure
distribution were also realized by the addition of a turning vane in the
primary nozzle. Figure 6-35 presents a comparison plot of P_{T_N} versus
span for the K = 1.17 Coanda with and without the vane. The installation
of the primary nozzle turning vane may be seen in Figure 6-36.

6.4.8   Phase III - Internal Radius Evaluation

6.4.8.1 General - Phase III testing was conducted to identify the effect
of reducing the internal Coanda nozzle radius of approach on the external
Coanda flow quality and performance. Design studies which were conducted
in parallel with this test program indicated that some improvements in
augmenter packaging could result from reductions of the internal Coanda
radius ratio (R_i/t). Up to this point in the test program the internal
radius of approach had been held constant at a value of R_i/t = 4.0.
Previous test data evaluating variations in internal radius ratio
indicated that reductions in internal radius to values below R_i/t = 3.0
could result in distortions in the Coanda nozzle pressure profile which
often produced external Coanda flow separation. Since further evaluation
of radius ratios less than 3.0 would be counterproductive, an internal
radius ratio of 3.0 was installed in the external Coanda K = 1.17 log
spiral configuration. See Figure 6-37 for comparison of shapes.

6.4.8.2 Turning Efficiency and Jet Growth - Turning efficiency and jet
growth are presented in Figure 6-38 and 6-39. Data obtained from testing
this configuration are compared with its R_i/t = 4.0 counterpart in
Figure 6-40, external Coanda jet total pressure decay, and Figure 6-41,
Coanda total pressure profile jet growth. Flow visualization for the two
radii is presented in Figure 6-42. Results of these comparisons indicate
only a slight variation in external Coanda flow quality was produced by
reducing the internal radius ratio from 4.0 to 3.0.
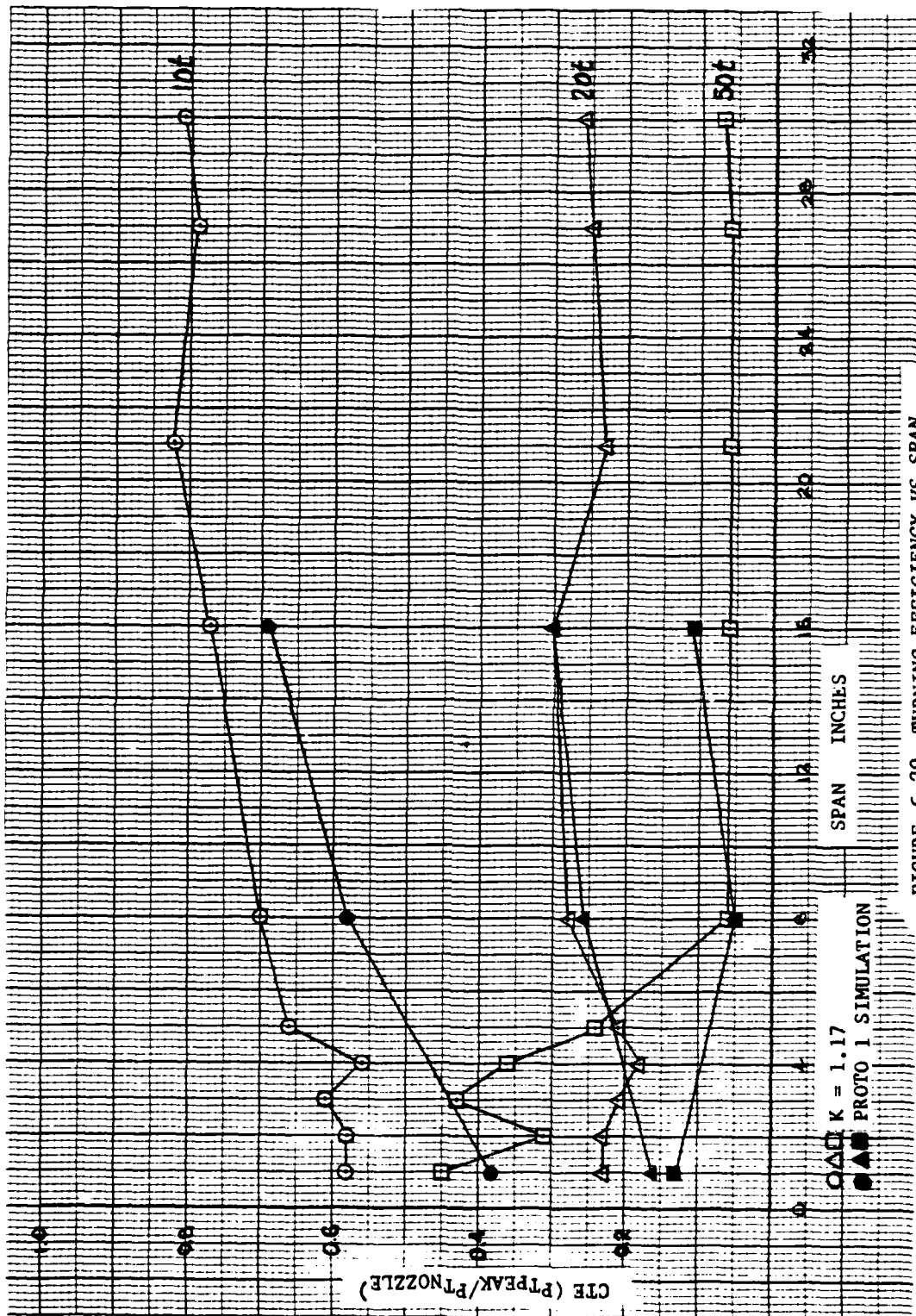
FIGURE 6-29 TURNING EFFICIENCY VS SPAN
K = 1.17 AND PROTO 1 SIMULATION

O △ □ K = 1.17
● ▲ ■ PROTO 1 SIMULATION

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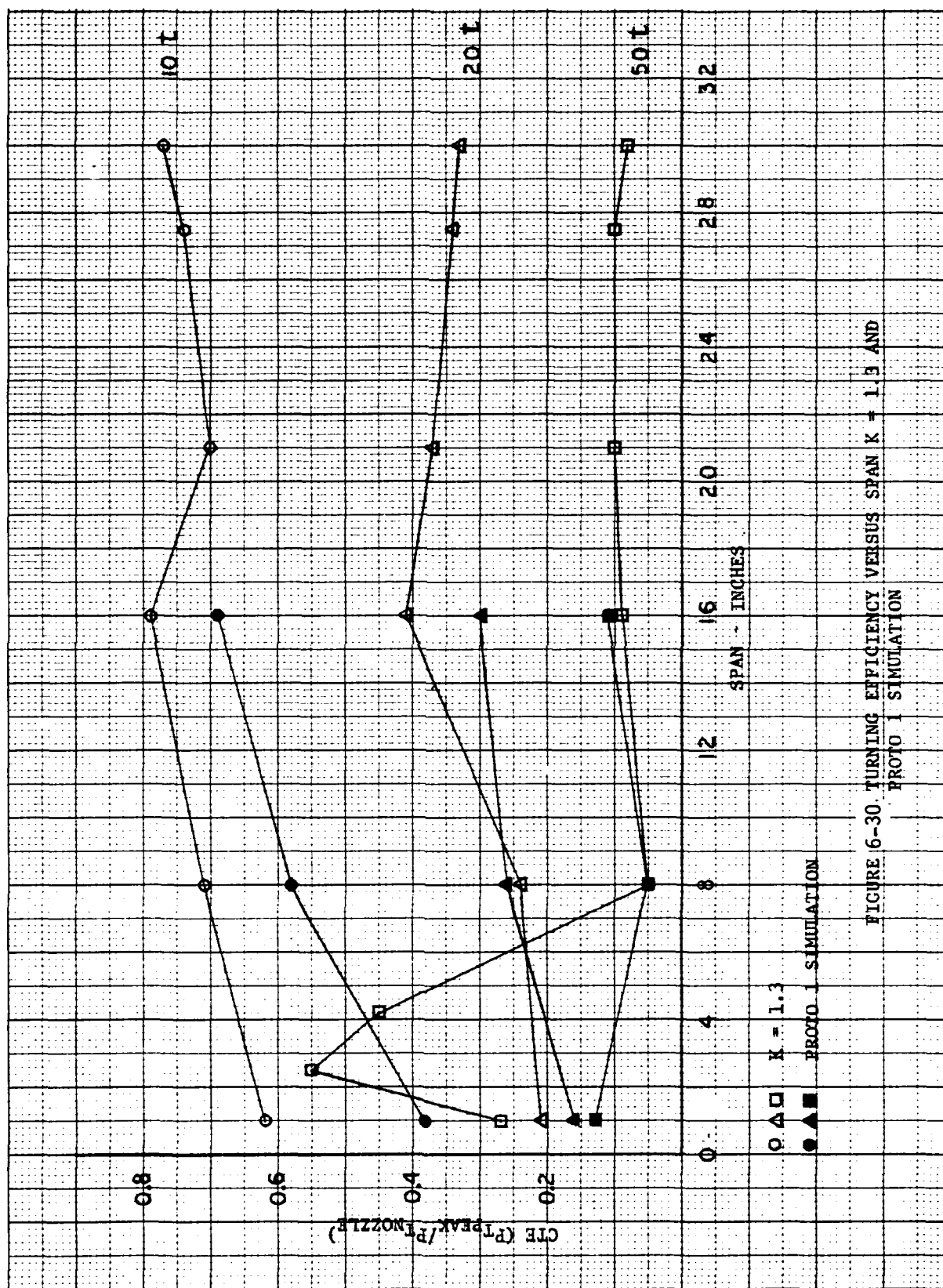
FIGURE 6-30. TURNING EFFICIENCY VERSUS SPAN K = 1.3 AND PROTO 1 SIMULATION

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RUN 52 - NO VANES
RUN 66 - ONE VANE
RUN 78 - TWO VANES
K = 1.3
DUMP DOOR CLOSED
NOMINAL PRESSURE RATIO = 2.05
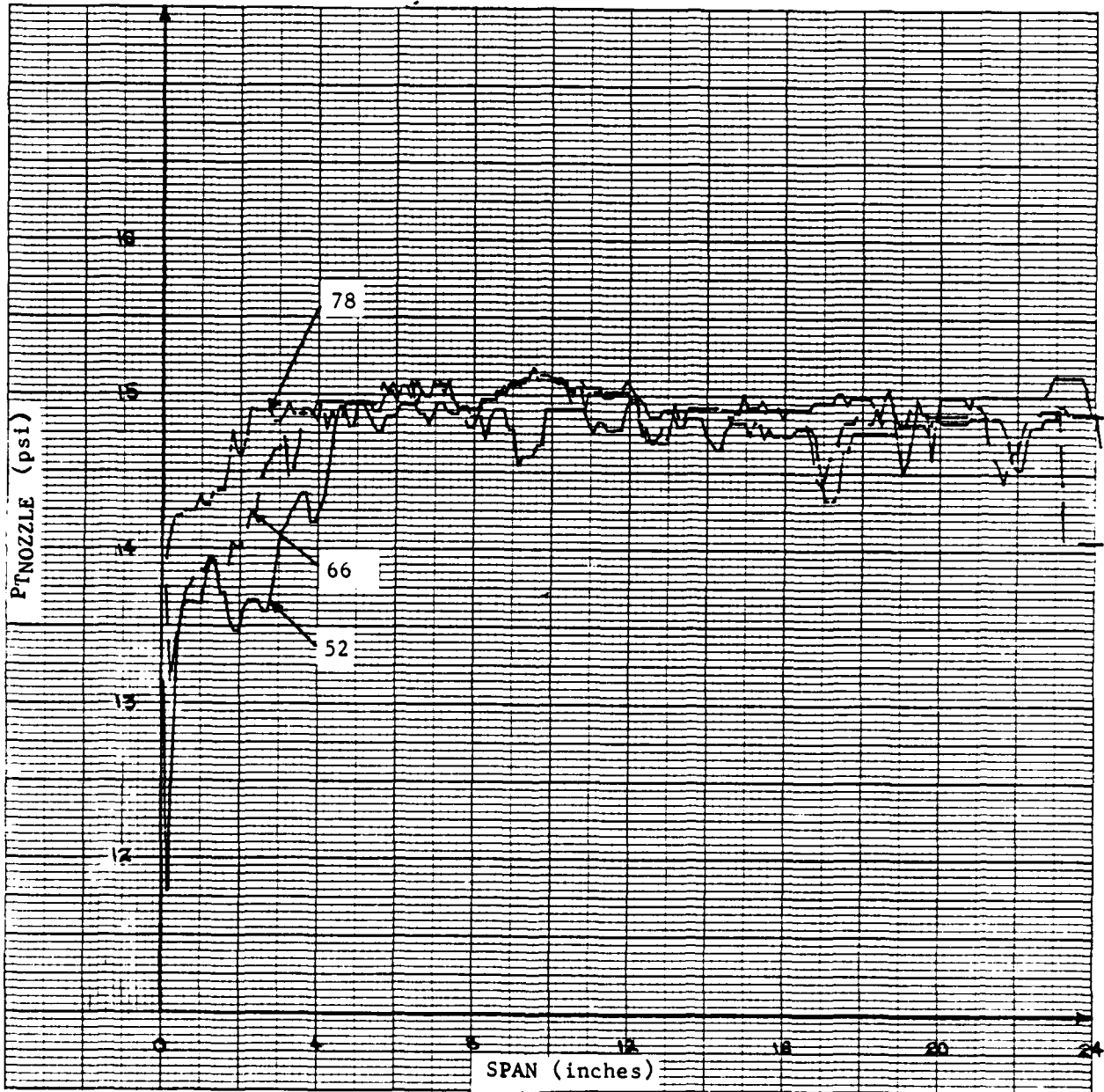
Figure 6-31   Nozzle Total Pressure Distribution Versus Span
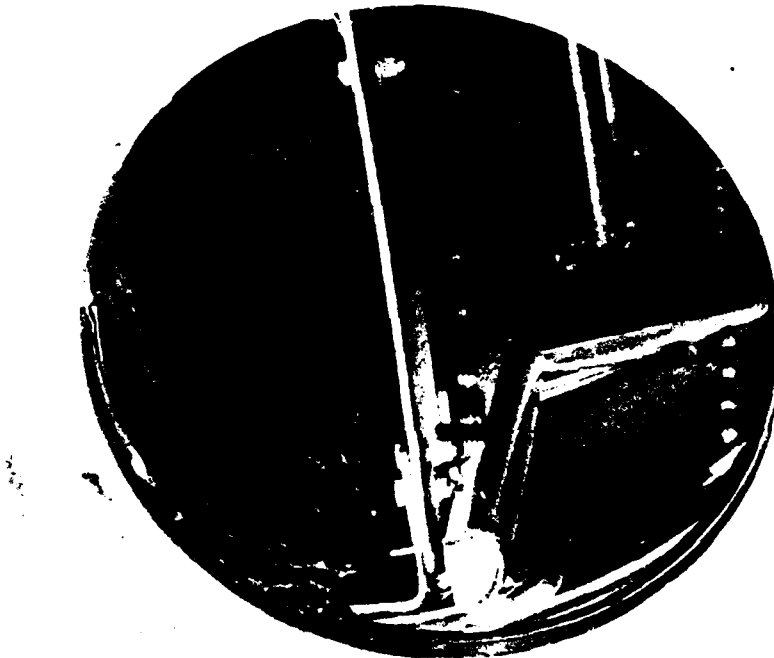
FIGURE 6-32. DEDICATED NOZZLE VANES

SINGLE VANE

FIGURE 6-33. DEDICATED NOZZLE VANES

DOUBLE VANE
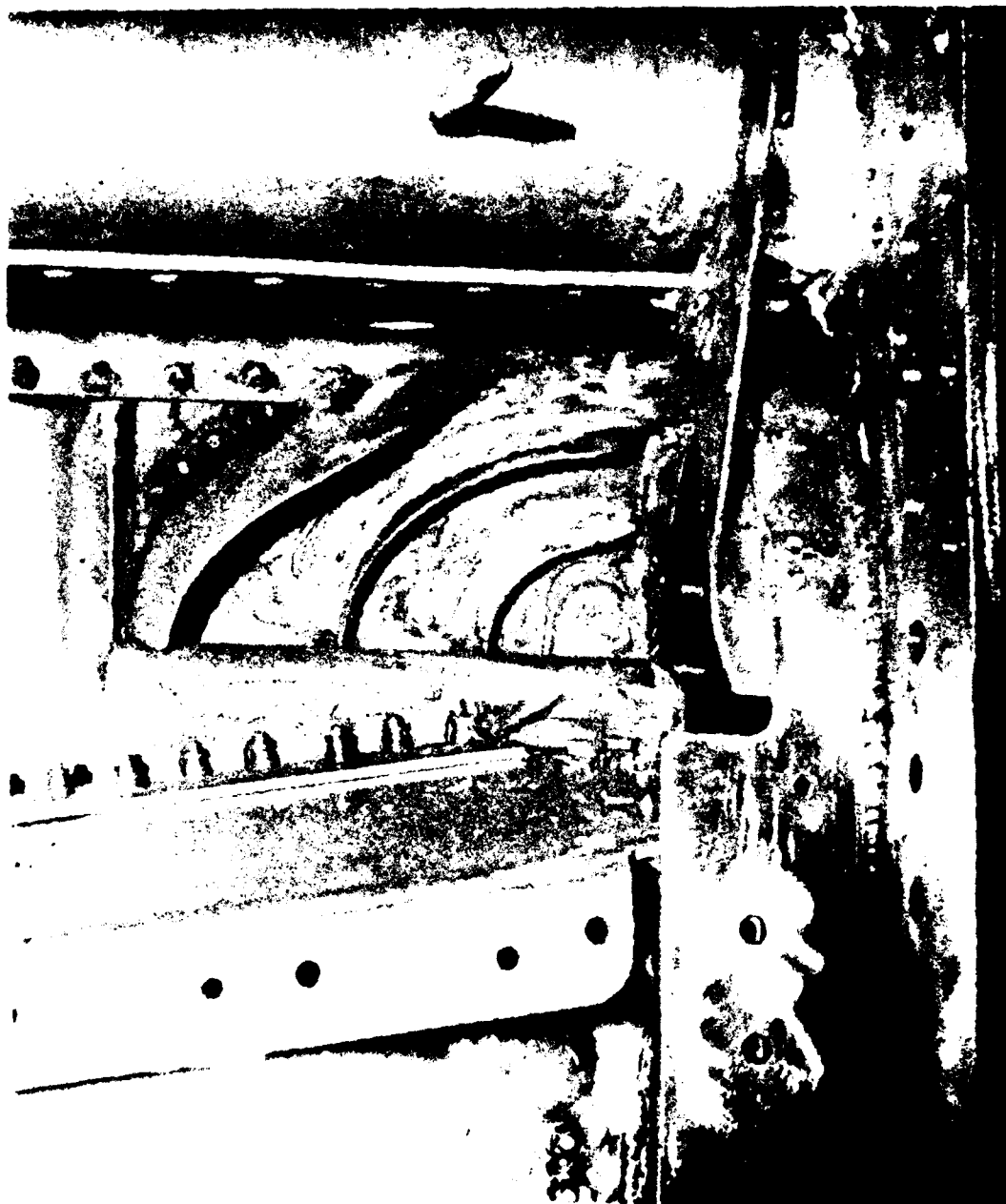
FIGURE 6-34   DEDICATED NOZZLE VANES

3 VANE CASCADE

RUN 82 - BASELINE CONFIGURATION
RUN 90 - TURNING VANE IN PRIMARY
                    NOZZLE
NOMINAL PRESSURE RATIO = 2.05
K = 1.17
DUMP DOOR CLOSED
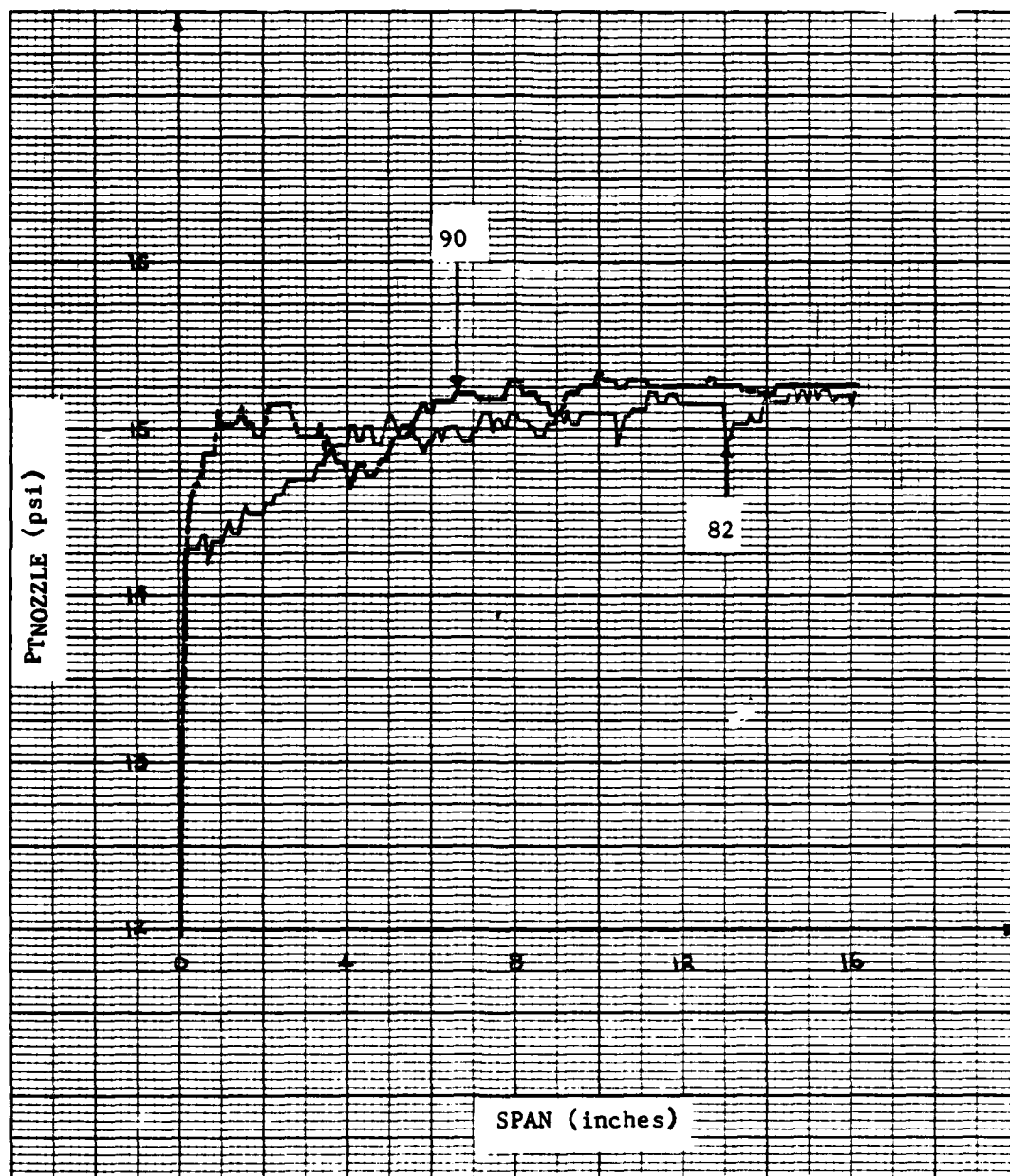
Figure 6-35   Nozzle Total Pressure Distribution Versus Span

FIGURE 6-36 .   PRIMARY NOZZLE VANE
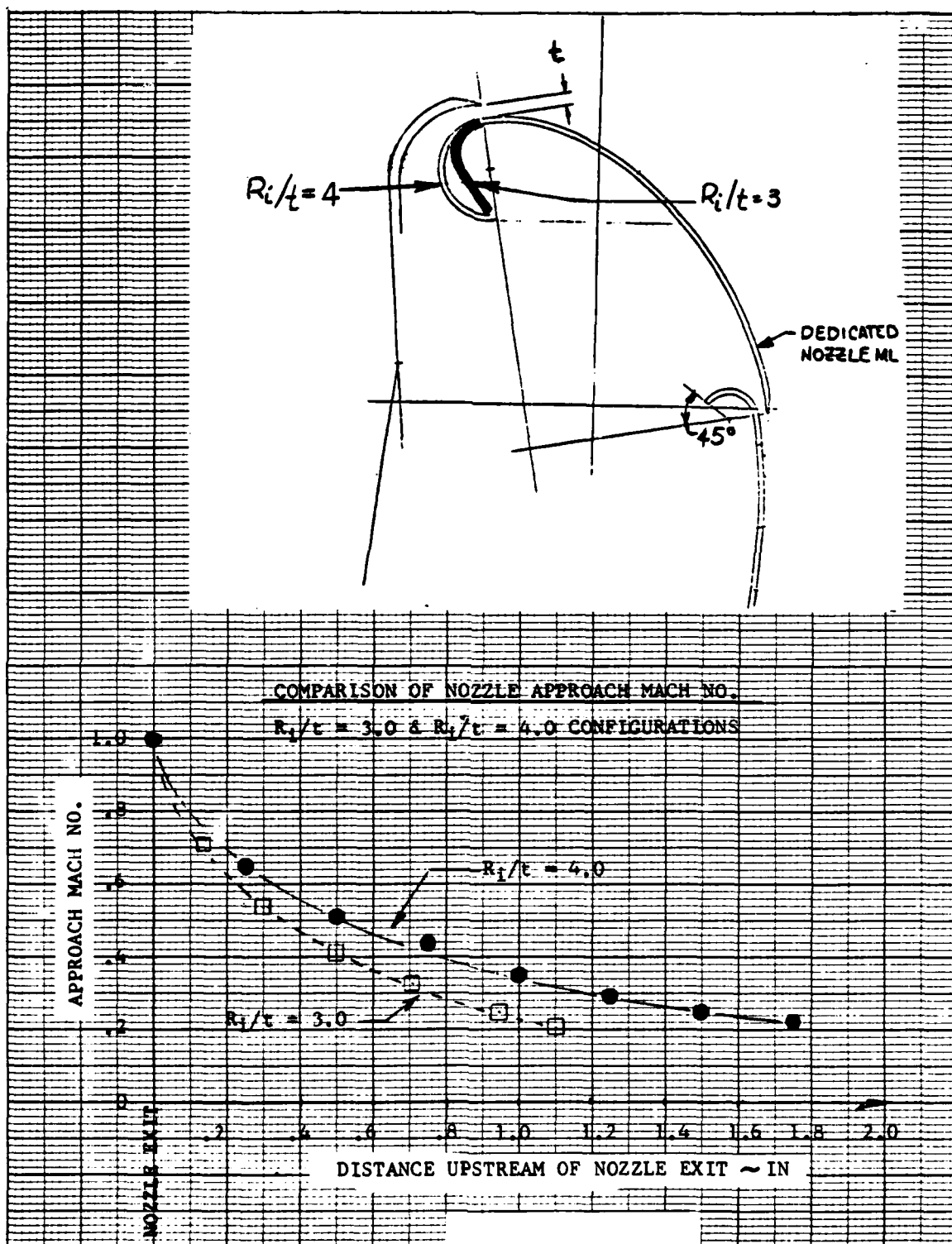
Figure 6-37 Sketch of Internal Radius Variations

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TURNING EFFICIENCY VS SPAN

FIGURE 6-38.

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JET GROWTH VS CHORD DISTANCE

SPAN STA 16 IN   INLET MACH NO. ≈ .25 DUMP 50%

K = 1.17 EXTERNAL SHAPE, R_L/t = 3.0

JET WIDTH (w/t)

CHORD LOCATION (s/t)

FIGURE 6-39

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COMPARISON OF TURNING EFFICIENCY FOR $R_1/t = 3$ & $R_1/t = 4$

$$CTE = \frac{P_{T_{PEAK}}}{P_{T_{NOZ}}} \times 100$$
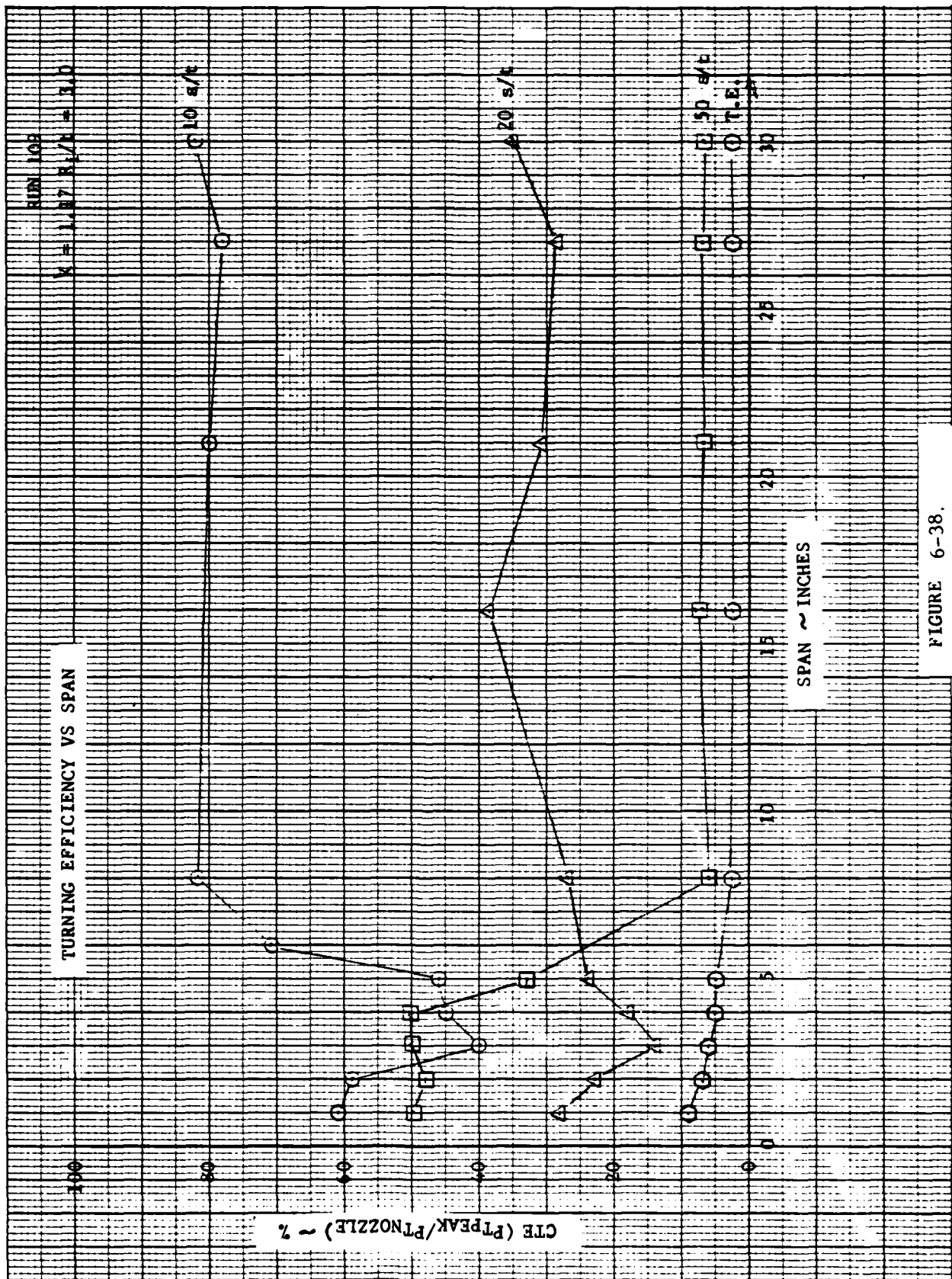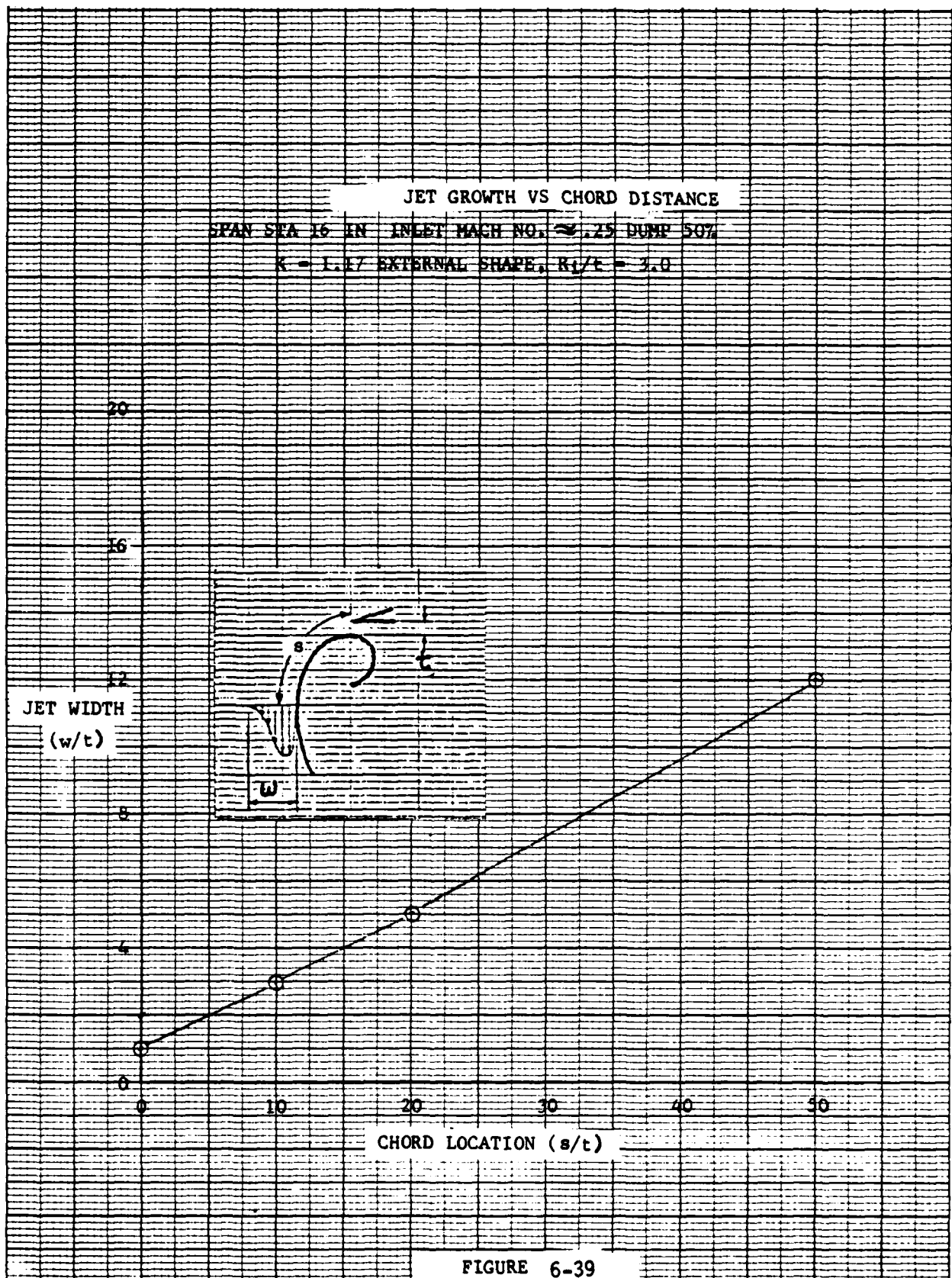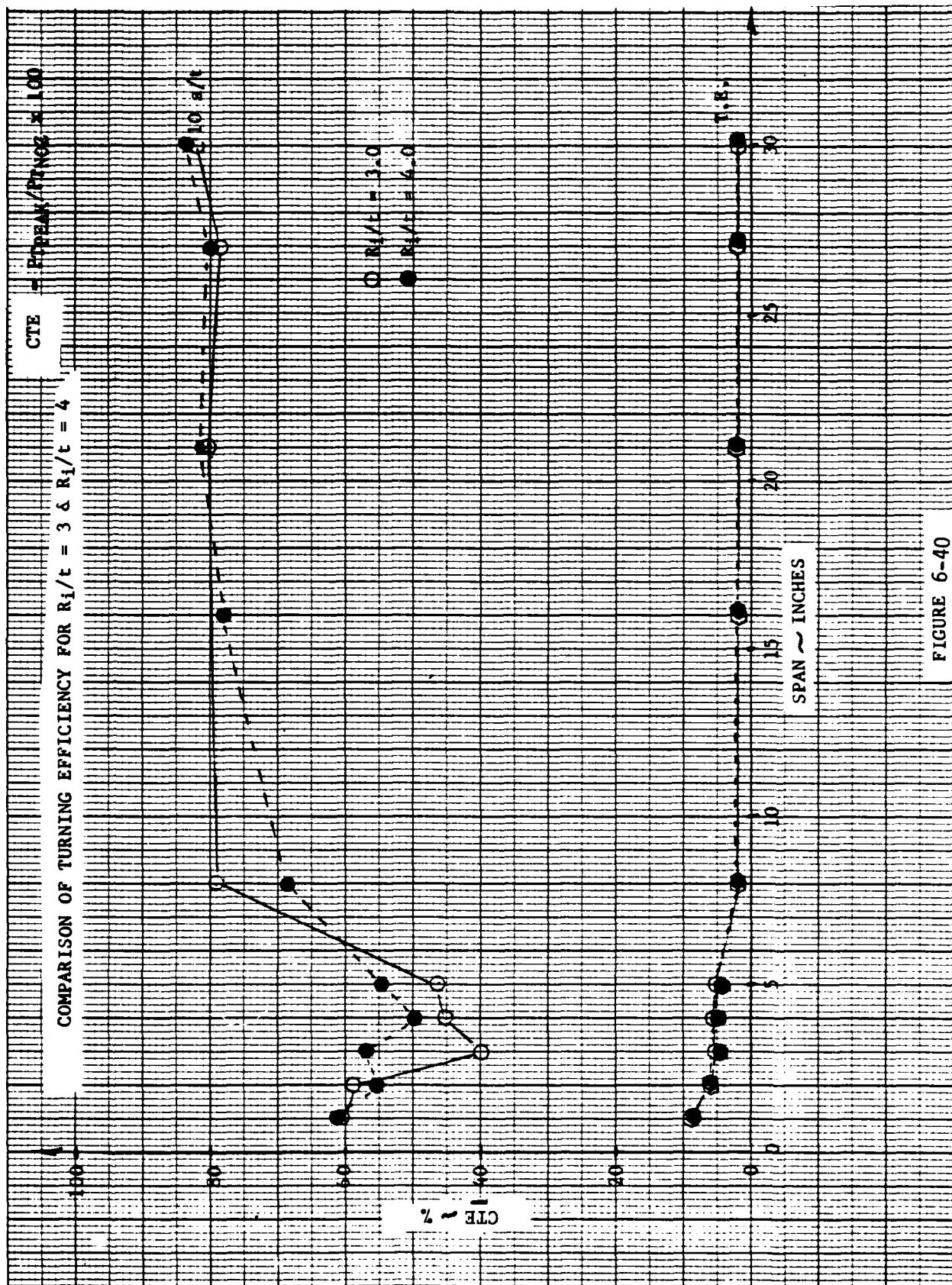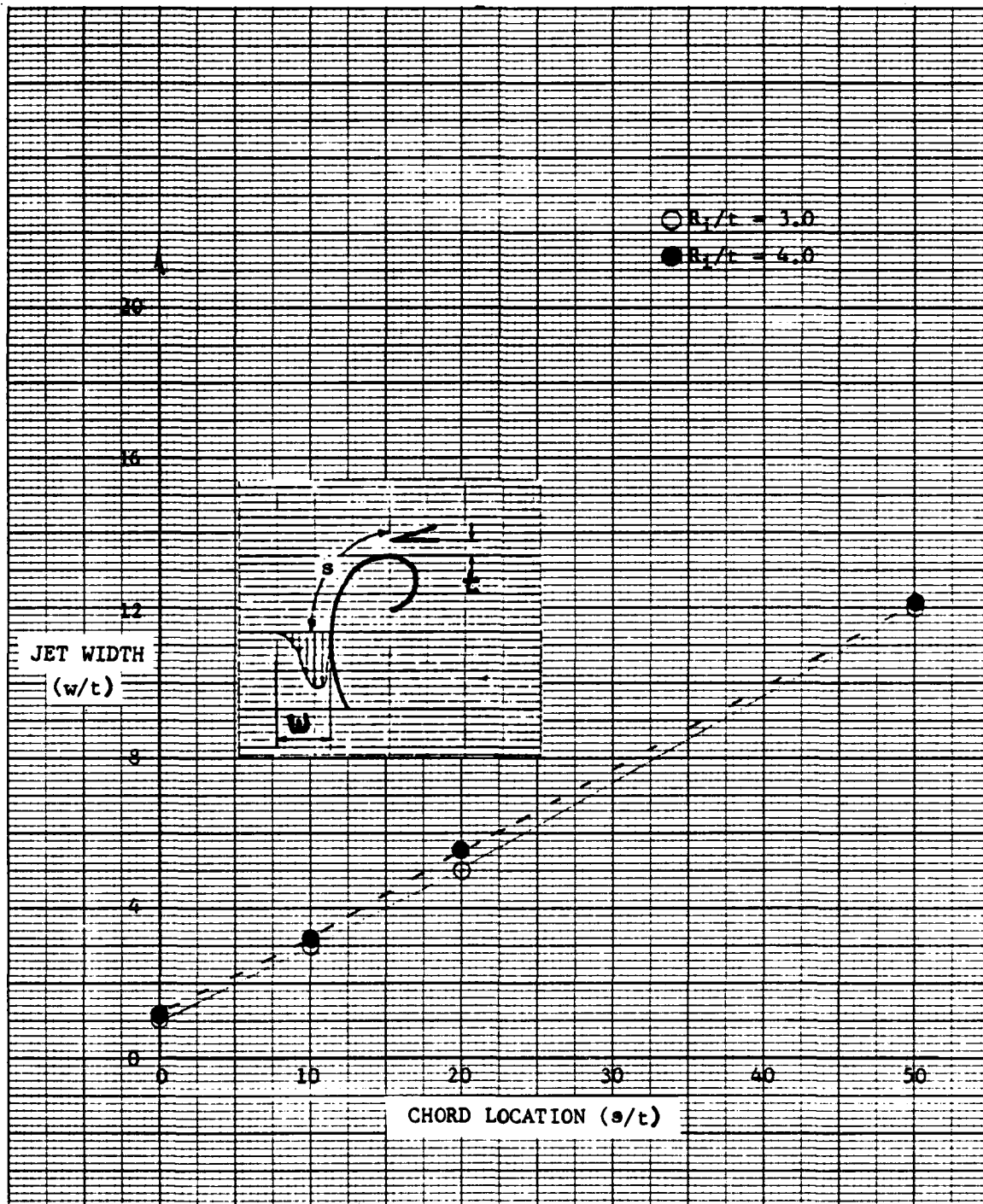
FIGURE 6-40

Figure 6-41 Jet Growth Comparison - $R_1/t = 3.0$
and $R_1/t = 4.0$ Configuration
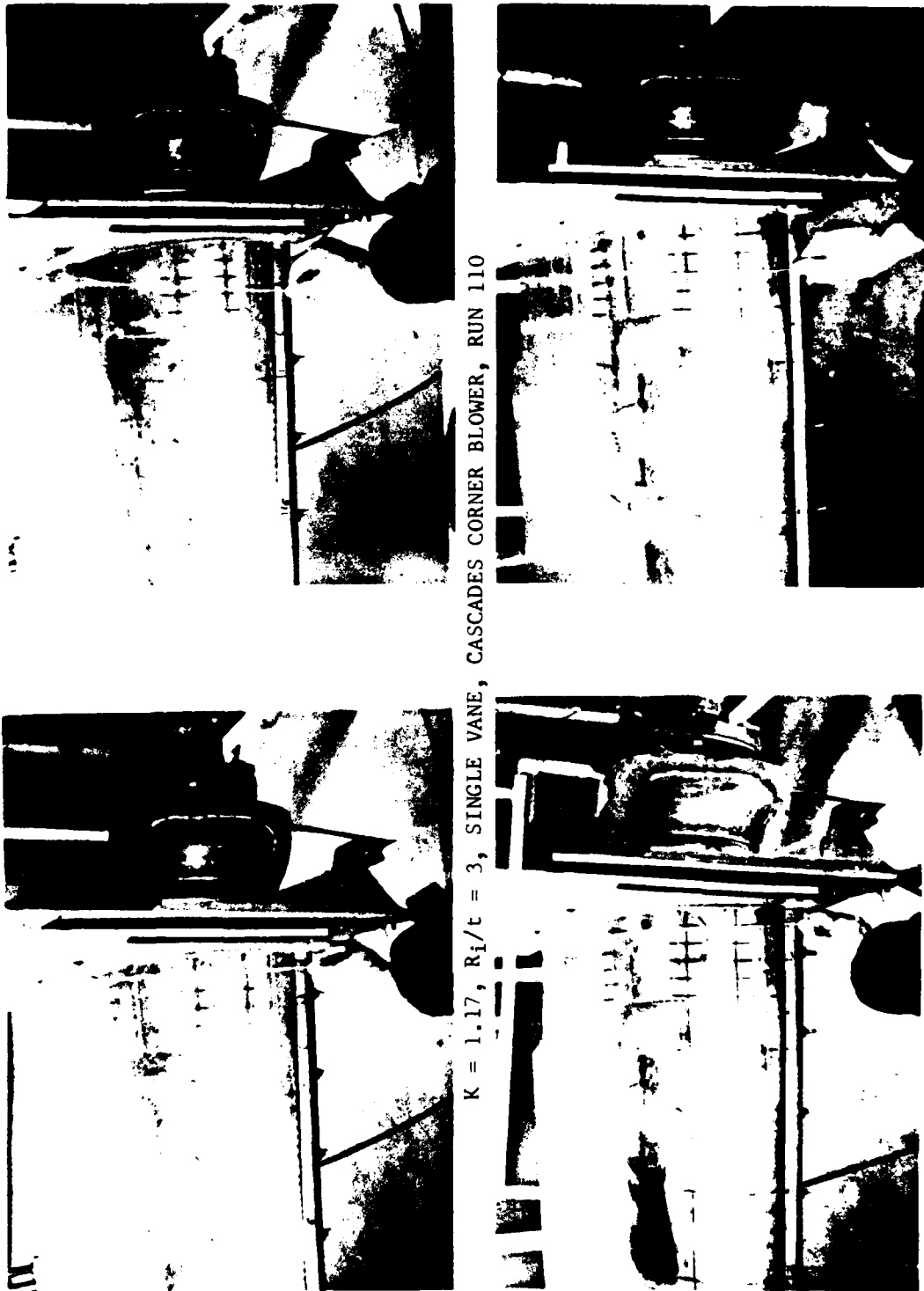
K = 1.17, R_i/t = 3, SINGLE VANE, CASCADES CORNER BLOWER, RUN 110

K = 1.17, R_i/t = 4, SINGLE VANE, CASCADES CORNER BLOWER, RUN 95

FIGURE 6-42 FLOW VISUALIZATION

6.4.8.3 <u>Flow and Velocity Coefficients</u> - Although no external Coanda flow separations were noted as a result of the reduction in internal radius ratio, slight reductions in nozzle flow coefficient were noted for the $R_1/t = 3.0$ configuration while velocity component coefficients were approximately the same for the two configurations (see Figure 6-43). It is believed that the slight reduction in flow coefficient is the result of the smaller internal radius distorting the nozzle exit pressure profile from the classical "top hat" mean profile normally measured on the larger internal radius configuration. Previous experience with distorted nozzle exit pressure profiles indicates that profiles of this type are often more prone to external Coanda flow separation and should be avoided.

6.4.9   Summary

Results of these tests satisfied the test objectives in providing definition for the improvement of the XFV-12A elevon. Specific conclusions are summarized below:

1   The WPAFB test facility is satisfactory for evaluation of flow characteristics and incremental performance changes of augmenter component models.

2.   The elevon segment model is a valid simulation of the full scale elevon for investigation of flow problems and evaluation of solutions.

3.   Relocating the elevon feed duct below the Coanda nozzle plane improves inboard flow characteristics.

4.   Minimal improvement of elevon flow characteristics is realized by increasing feed inlet flow area.

5.   The external Coanda shape defined by K = 1.3 resulted in the highest turning efficiency in the region immediately downstream of the nozzle when compared with the K = 1.17 and Proto 1 shapes. The velocity coefficient for the K = 1.3 shape was superior to the one for K = 1.17 and comparable to Proto 1.

6.   The addition of properly placed vanes in the inboard end of the primary nozzle and in the dedicated nozzle improved inboard primary nozzle pressure distribution and inboard flow quality.

7.   Results of tests with a reduced internal Coanda nozzle radius were largely inconclusive. However, the slight reduction in flow coefficient obtained for the smaller radius is of sufficient concern to defer adoption.

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FLOW AND VELOCITY COEFFICIENTS

$R_i/t = 3.0$ & $R_i/t = 4.0$ CONFIGURATIONS

$C_D = \dfrac{\text{VENTURI MEAS. FLOW}}{\text{IDEAL NOZZLE FLOW}}$

$C_V = \dfrac{\text{LIFT}}{\text{NOZZLE ISENTROPIC THRUST}}$

$R_i/t = 3.0$

$R_i/t = 4.0$

C_D VS NPR

C_V VS NPR
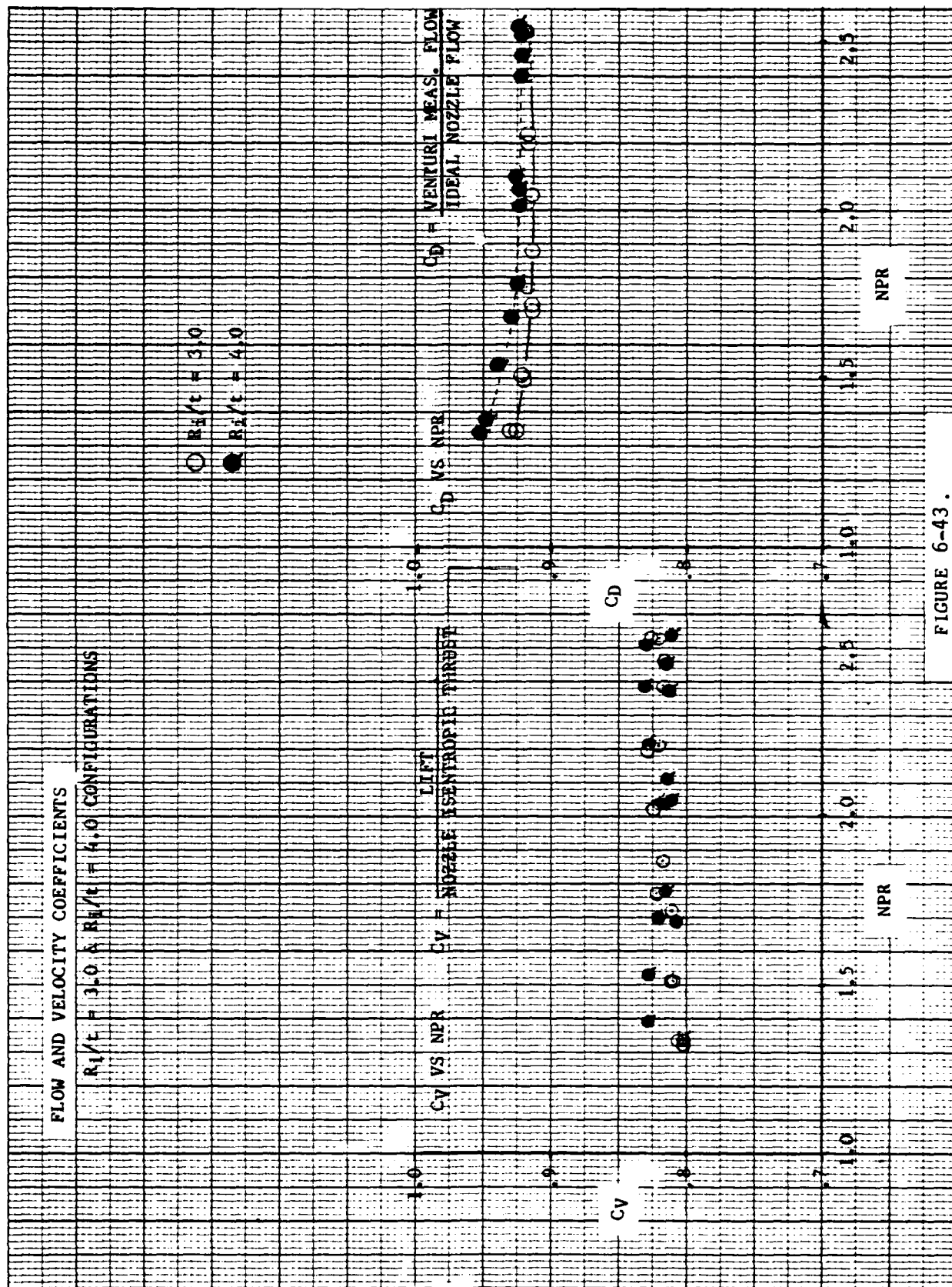
FIGURE 6-43.

7.0 CENTERBODY NOZZLE DEVELOPMENT

7.1   GENERAL

The process of ejector thrust augmentation is driven by the primary jet entrainment, which pumps the secondary stream through the ejector. Thus, increasing the jet entrainment rate will increase the thrust augmentation. The purpose of this section is to summarize the research performed in order to develop a nozzle for increasing the entrainment of the central jet in the XFV-12A wing augmenter. First, use of the jet mixing analysis for optimization of the basic hypermixing nozzle will be described. This will be followed by a description of progress in the development of an asymmetric cross slot nozzle. Finally, a study of the symmetric cross slot nozzle will be presented. It is shown that the use of a cross slot nozzle and a corresponding increase of ejector inlet ratio increases the thrust augmentation of the wing ejector from 1.46 to 1.64.

7.2   HYPERMIXING NOZZLES

Significant increases in thrust augmentation have been achieved with the hypermixing nozzle (Reference (23) and (24)) shown in Figure 7-1. The alternating jet segments at the exit of the nozzle serve to introduce a row of streamwise vortices into the jet. These vortices enhance the turbulent entrainment, and thus pump additional fluid through the ejector. The rate of entrainment depends on the strength and spacing of the vortices which are functions of the initial deflection and size of the jet segments.
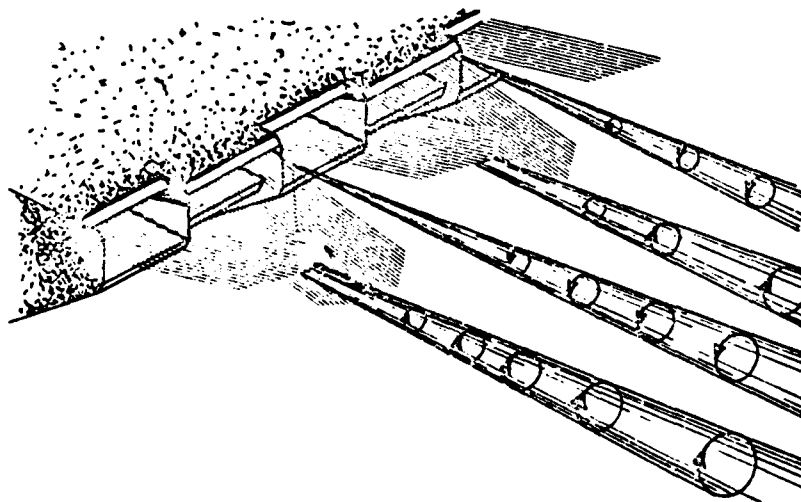
Figure 7-1 Hypermixing Nozzle Exit

The combination of these parameters which gives the best performance depends on the ejector inlet area ratio and lengths, as well as the nozzle location. On the basis of the limited data available, a nozzle with an initial deflection angle of 7° and a segment aspect ratio of 12.5 was chosen for the XFV-12A configuration tested at NASA Langley. As part of the present aircraft development program, the jet mixing analysis previously described was used to evaluate the possibility of improving performance by redesigning the hypermixing nozzle. The ejector section at the wing root was chosen for analysis.

The computed effect of varying the jet parameters is shown in Figure 7-2.
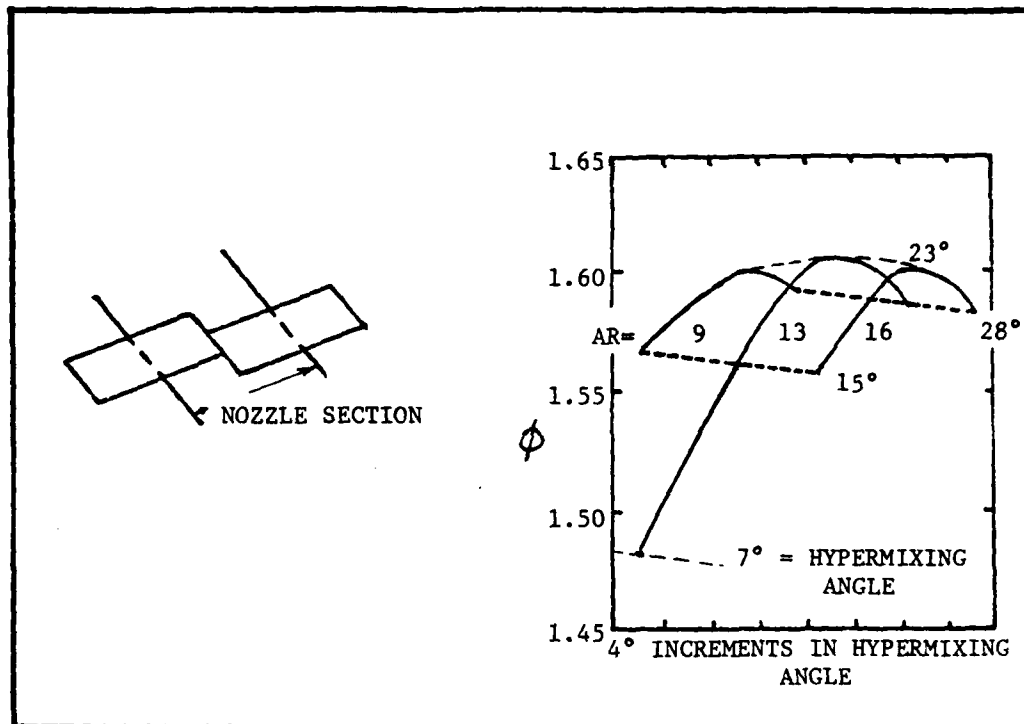
Figure 7-2 Effect of Jet Parameter Variations on Thrust Augmentation

Changes in the deflection angle are seen to have greater effect on the augmentation then the aspect ratio. In Figure 7-3, the jet deflection angle is held constant at 23°, and the aspect ratio is varied over the range from AR = 3 to AR = 24. The computed velocity distributions are as shown. In the figures, the hypermixing jet runs along the span in the foreground with a Coanda jet behind it; the other Coanda jet has been removed to permit a clear view of the hypermixing jet. At low aspect ratios, the vortices are close together so that they soon merge and cancel each other. As seen in the velocity distribution for the AR = 3 case, this limits the spreading of the jet, which then limits the augmentation.
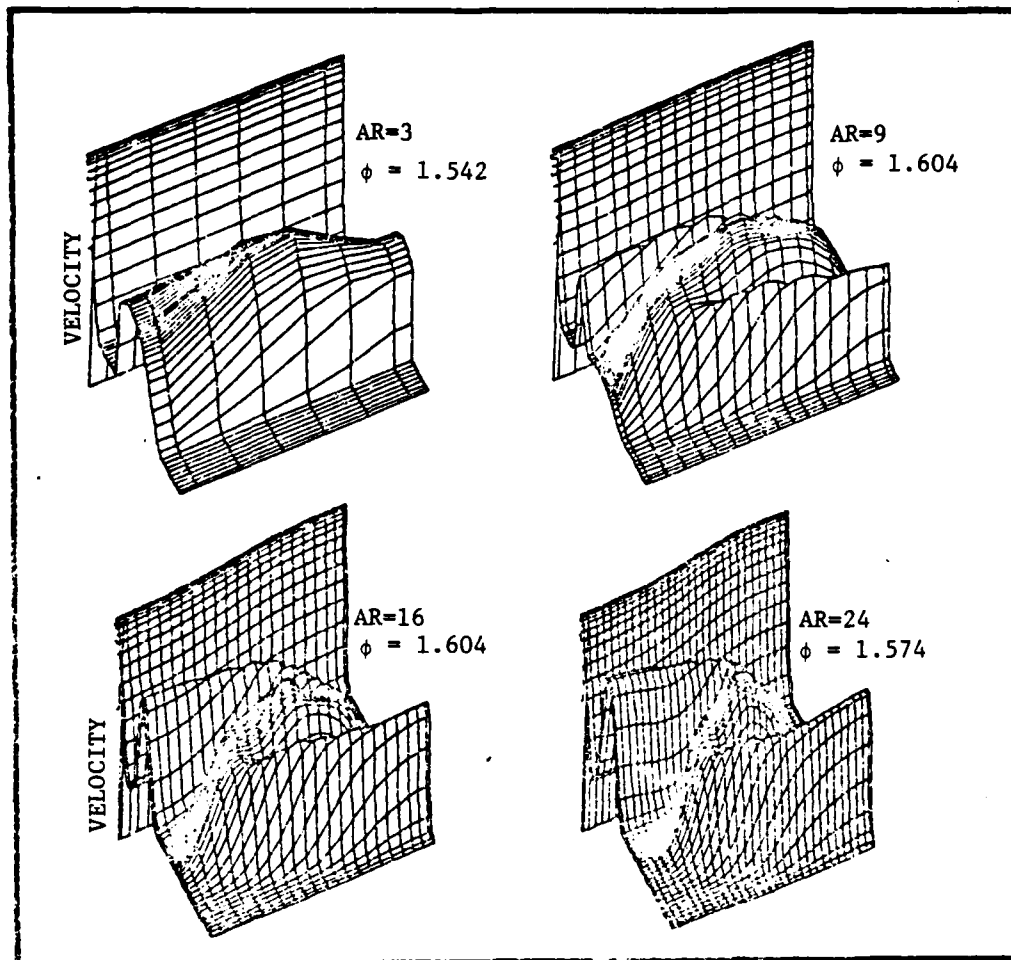
Figure 7-3 Computed Effect of Hypermixing Aspect Ratio on the Jet Mixing

As the aspect ratio is increased, the interference between vortices is
reduced, so that the mixing and augmentation also increase. In the limit,
as the aspect ratio becomes too large, the number of vortices is reduced
to the point that the mixing and augmentation decrease. This can be seen
in the velocity distributions, which are normalized with respect to the
vortex spacing. For the AR = 24 case, the hypermixing effect does not
extend to the ends of the segment shown.

The computed effect of aspect ratio on augmentation is shown for these
jets in Figure 7-4. The thrust augmentation has a relatively broad peak
for aspect ratios between 10 and 15. This is consistent with the
measurements of Salter (Reference (25), who found little change in

7-3

the entrainment of hypermixing jets over this range. Because the aspect
ratio of the baseline aircraft nozzle is near the middle of this range, no
additional effort was directed towards optimizing this parameter.

Figure 7-4   Effect of Hypermixing Nozzle Aspect Ratio On
Thrust Augmentation, Deflection = 23°

As seen in Figure 7-2, a significant increase in augmentation was
predicted for an increase in the hypermixing angle from 7° to 23°. In
order to test this prediction a set of four nozzles were built and tested
in a mid-span section model of the XFV-12A wing. In Figure 7-5, the
augmentation measured at a diffuser area ratio of two is compared to the
computed thrust increments. Although the augmentation did reach a maximum
at approximately the same angle as predicted, the measured thrust
increment was not as large. The measured and calculated velocity
distributions at the ejector exit, shown in Figure 7-6, were qualitatively
similar. However, there were quantitative differences which also
indicated that the analysis could be improved.

7-4
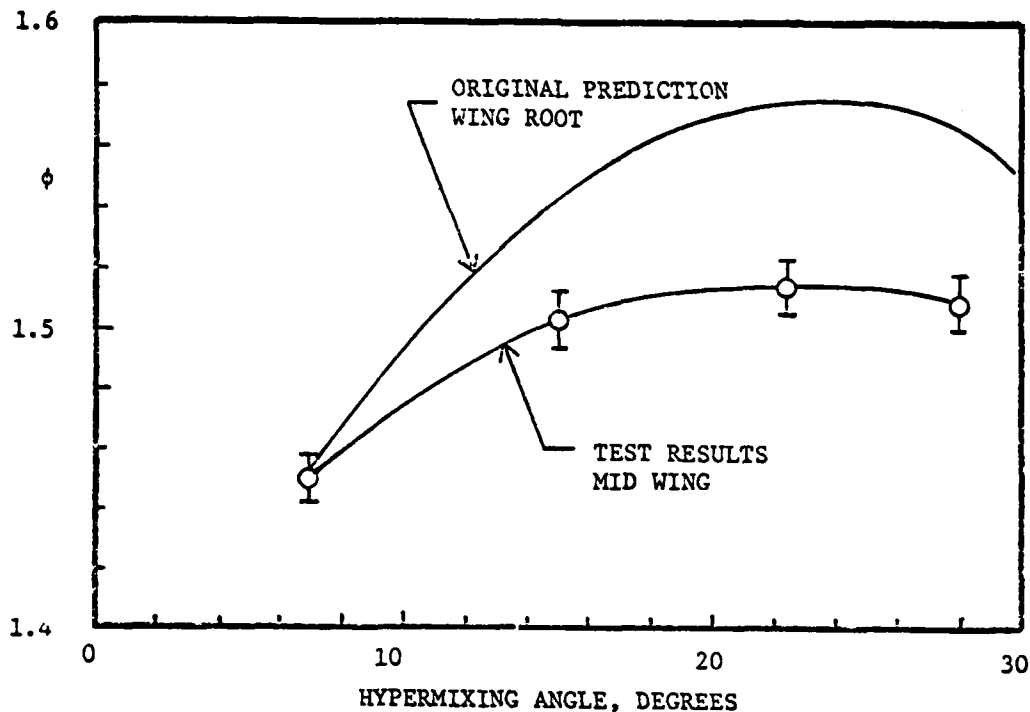
Figure 7-5 Comparison of Computed and Measured Effect of Hypermixing Angle

7.2.1 Analytical Improvements

Because data was not available, initial values of the secondary flow angularity and turbulence constants had to be assumed. In order to improve the specification of initial conditions, these parameters were measured for each of the four nozzles. The magnitude and direction of the secondary flow was measured with a five-port pitot-static probe at the positions shown in Figure 7-7. A vector diagram of the data measured at the center of a segment of the 15° nozzle is shown in Figure 7-8. Typical data is presented in Figure 7-9. The data from these nozzles were then generalized to improve the inlet angularity subroutine used in the jet mixing analysis.

Because it is difficult to measure turbulence parameters in supersonic jet flows, the effect of the turbulence constants on the jet mixing was studied parametrically. In Figure 7-10, the measured and computed profiles at the center and end of a jet segment are compared for the original values of these constants. Better agreement was obtained by increasing the kinetic energy by a factor of four, as shown in Figure 7-11, or by decreasing the dissipation constant by a factor of 1/16, as shown in Figure 7-12. Since the eddy viscosity is proportional to k^2/ε, these changes were equivalent to increasing the eddy viscosity by a factor of 16.
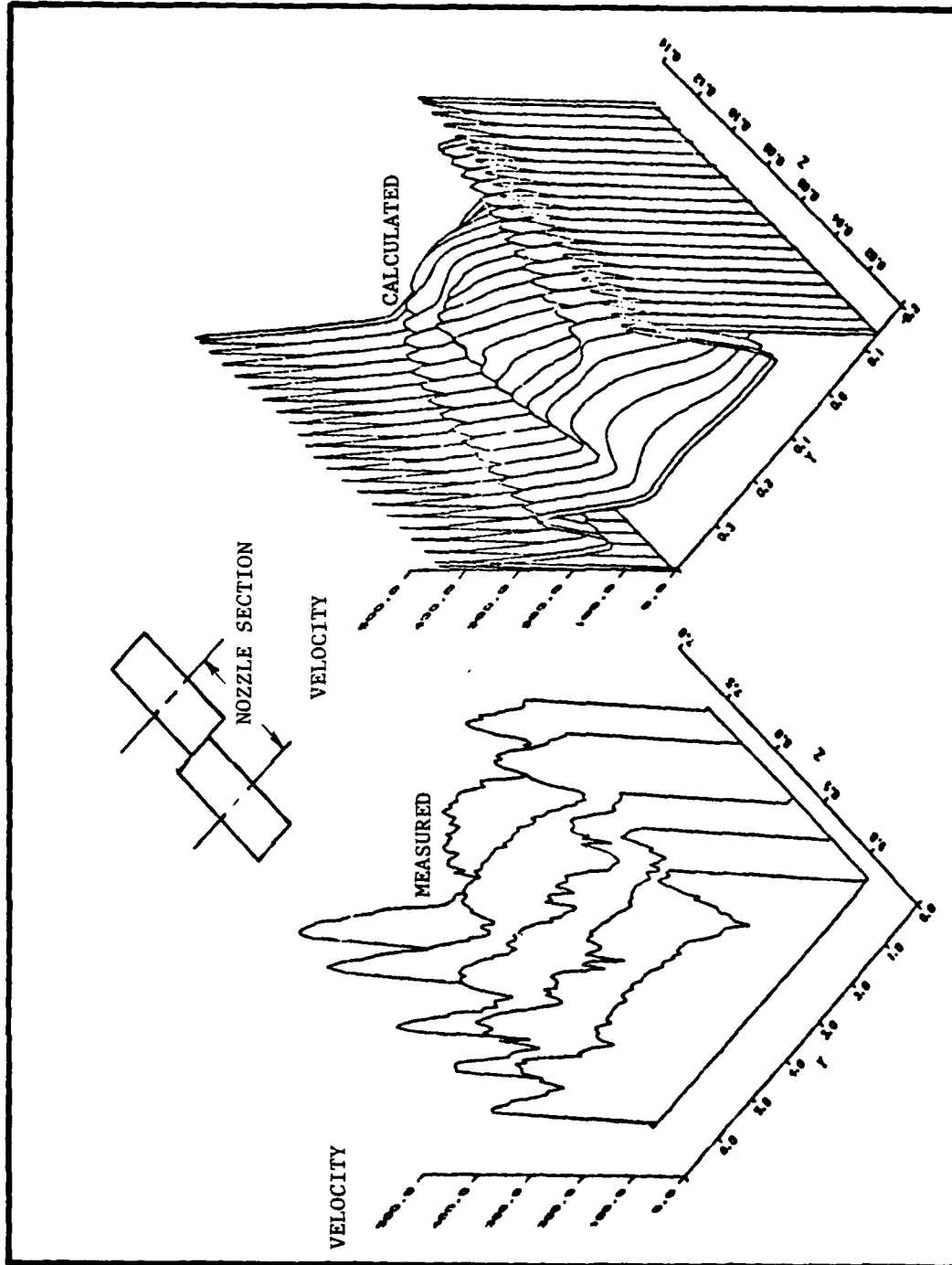
Figure 7-6 Comparison of Hypermixing Velocity Distributions at Ejector Exit
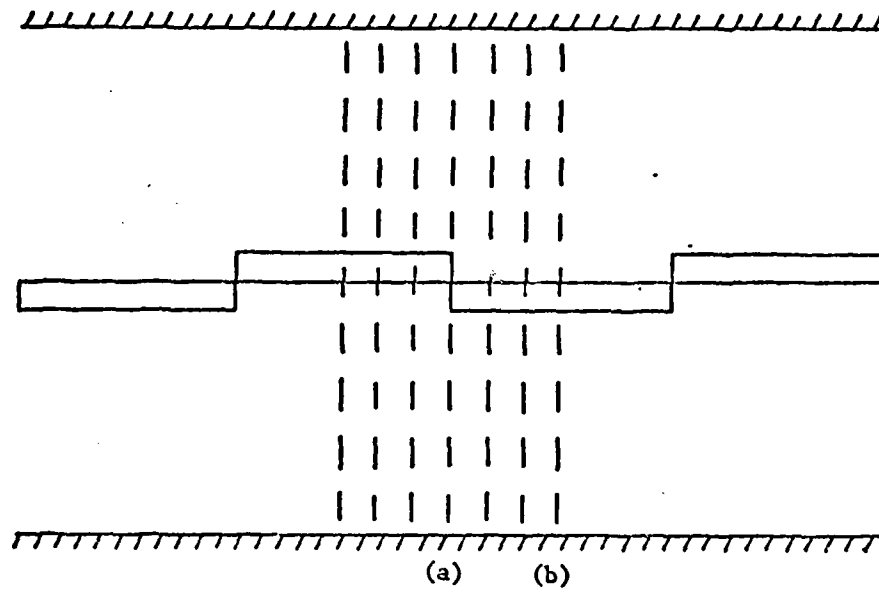
Figure 7-7 Hypermixing Nozzle Survey Stations
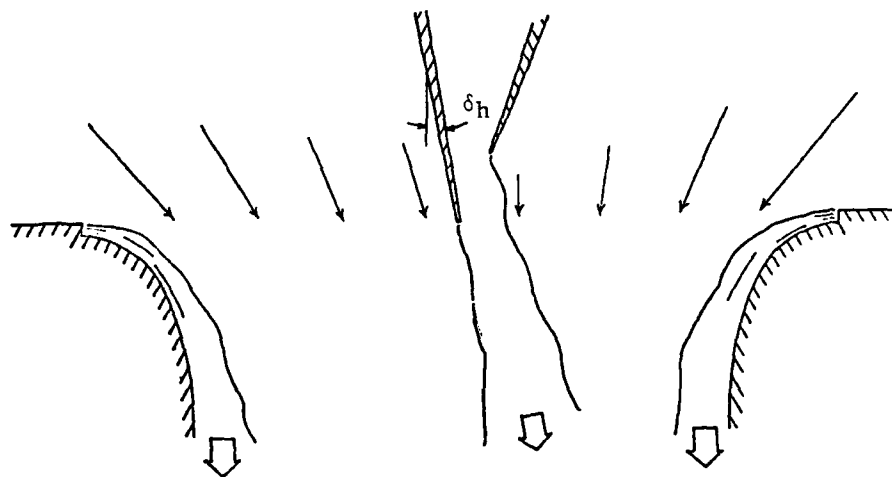
Figure 7-8 Measured Inlet Velocities

7-7
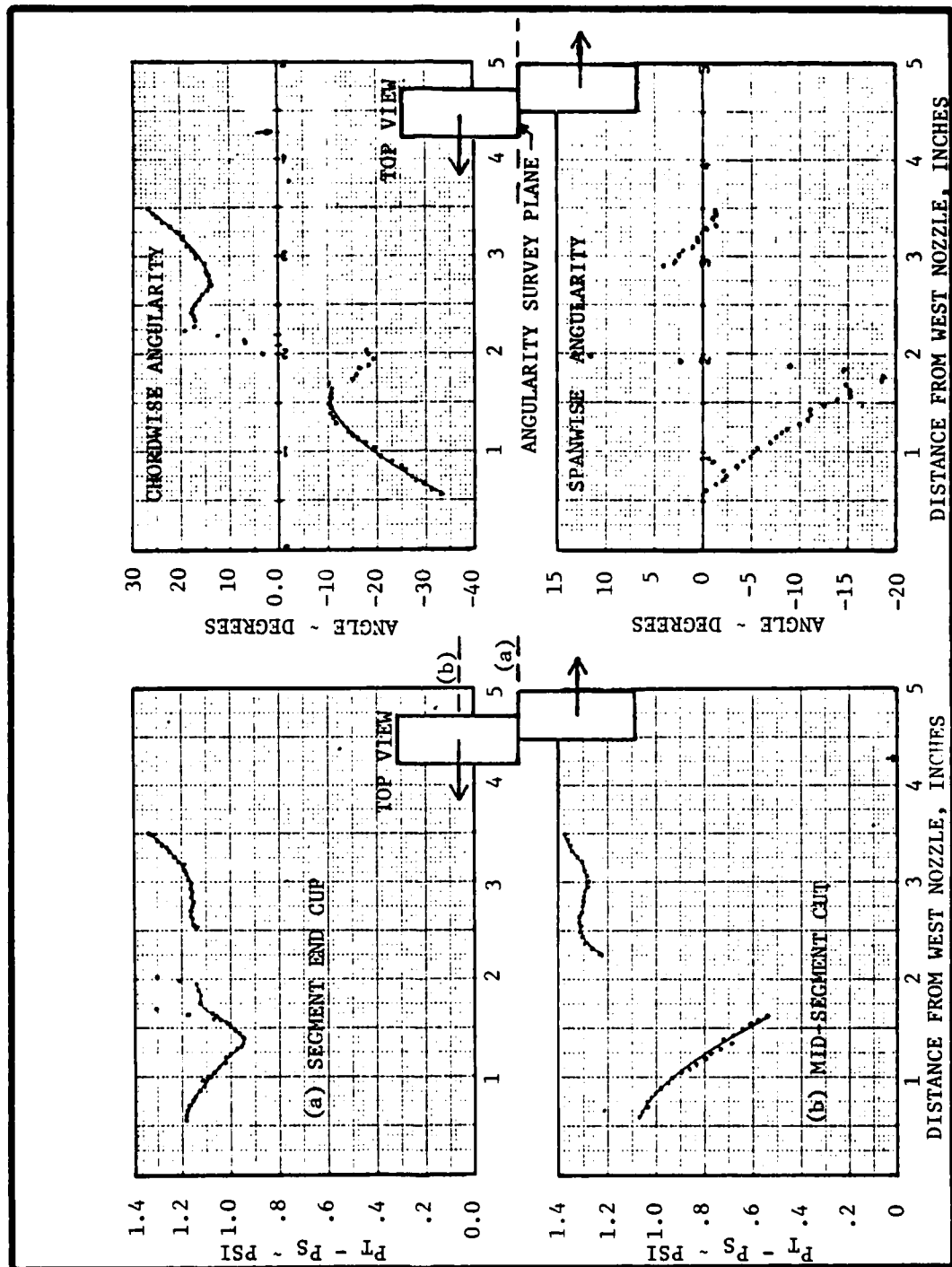
Figure 7-9. Dynamic Pressures and Angularity of Inlet Secondary Flow, 15° Hypermixing Angle and $A_2/A_0 = 15.2$
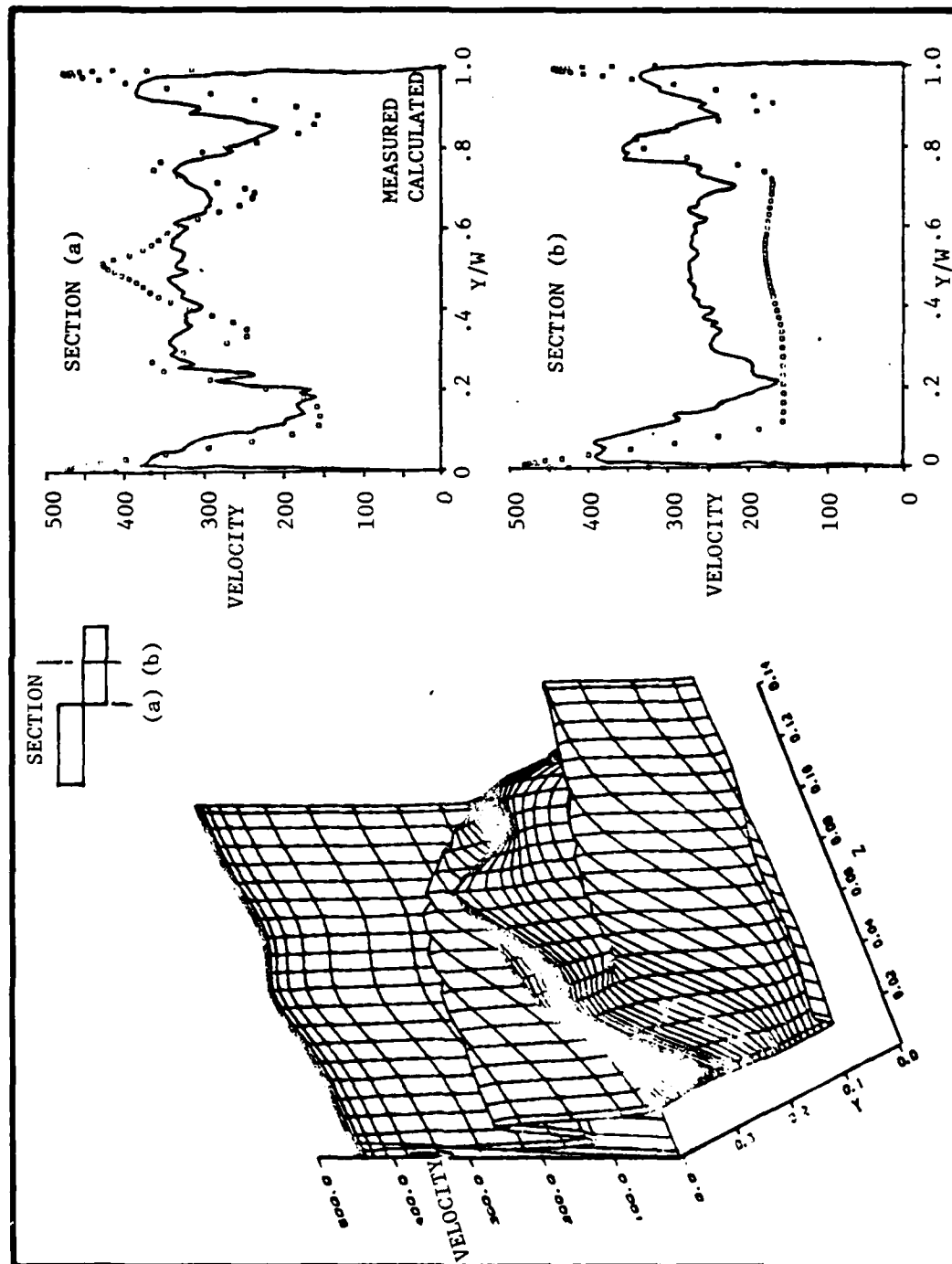
Figure 7-10 22.5° Hypermixing Jet Velocity Distribution with Original Values of C_k and C_ϵ
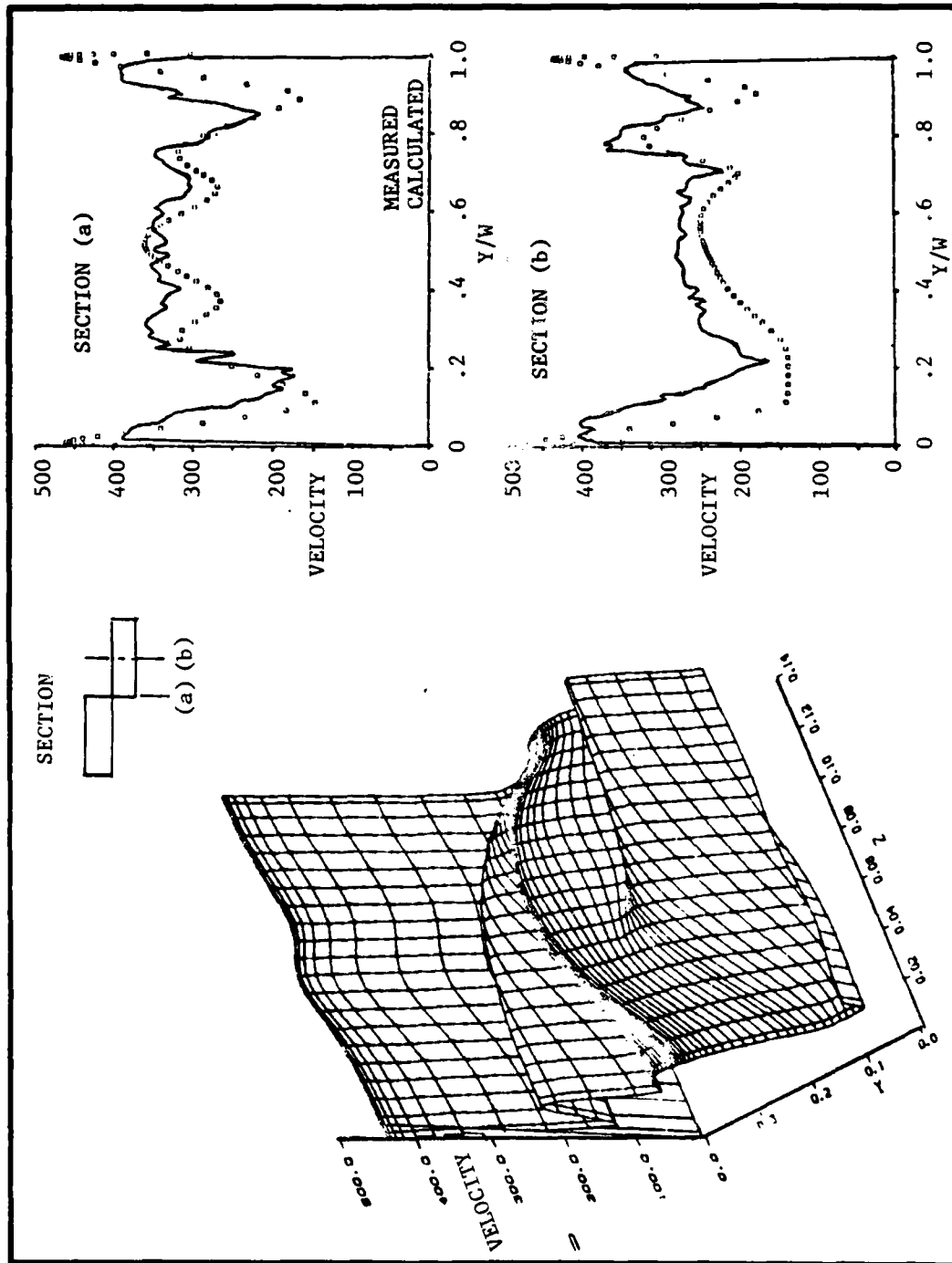
Figure 7-11   22.5° Hypermixing Jet Velocity Distribution with $C_k = 4. \times C_{k, ref.}$
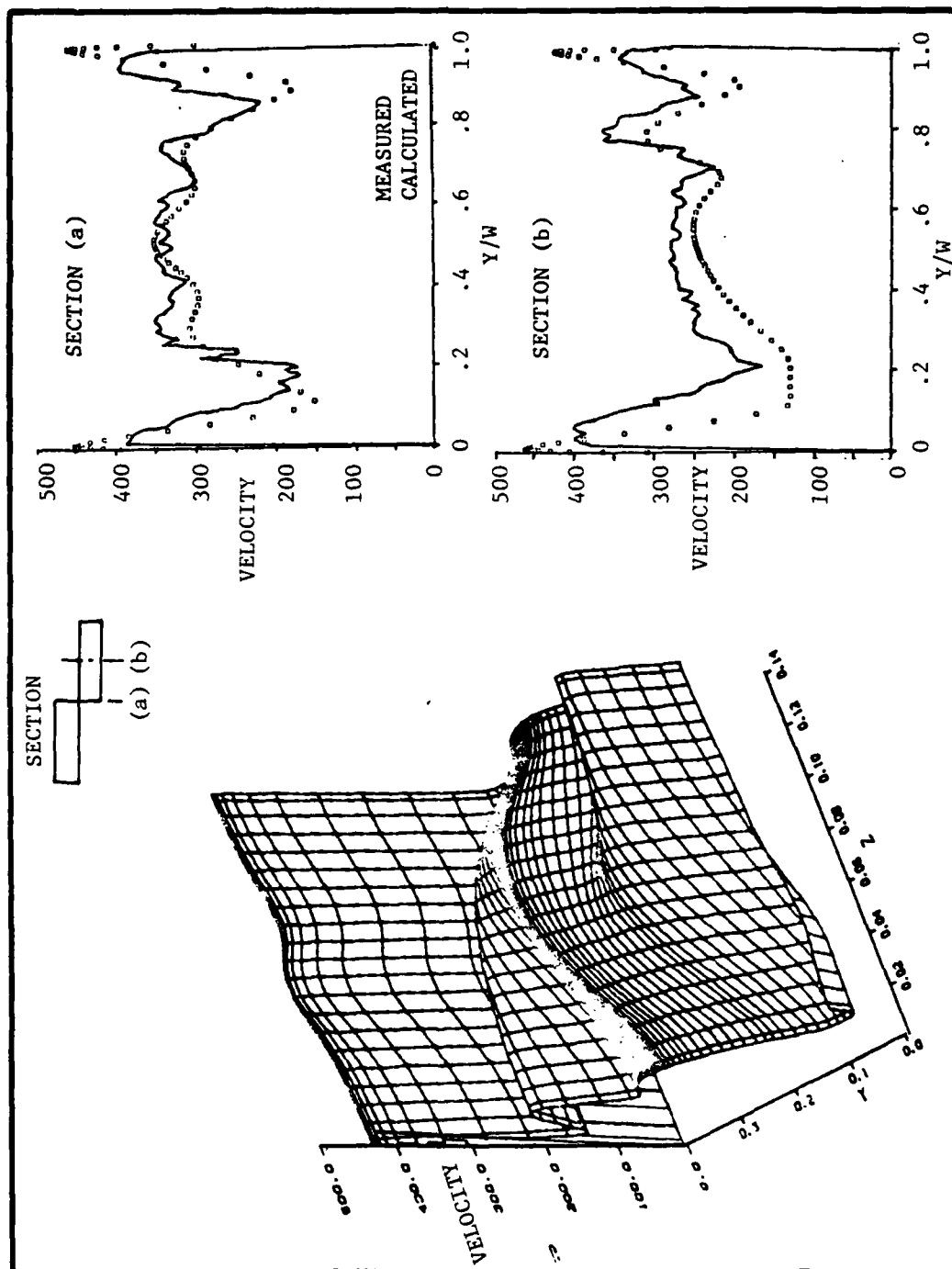
Figure 7-12 22.5° Hypermixing Jet Velocity Distribution with $C_\epsilon = C_{\epsilon,ref}/16$

It thus became necessary to measure the turbulence intensity in the jets in order to determine the correct values of the turbulence constants. Measurements made with hot film and kulite probes both indicated that a value of $C_k = 0.04$ was appropriate. The corresponding value of the dissipation constant was found to be $C_\epsilon = 0.003$. These values are typical of channel flows.

Figure 7-13 shows a comparison of the measured augmentation with the increments computed using these values of the constants and the improved angularity subroutine. Good agreement was obtained up to approximately 20°. It was felt that the difference at larger angles was due to violating the basic small angle assumption of the analysis, and to neglecting the effect of swirl on the turbulence production. In order to
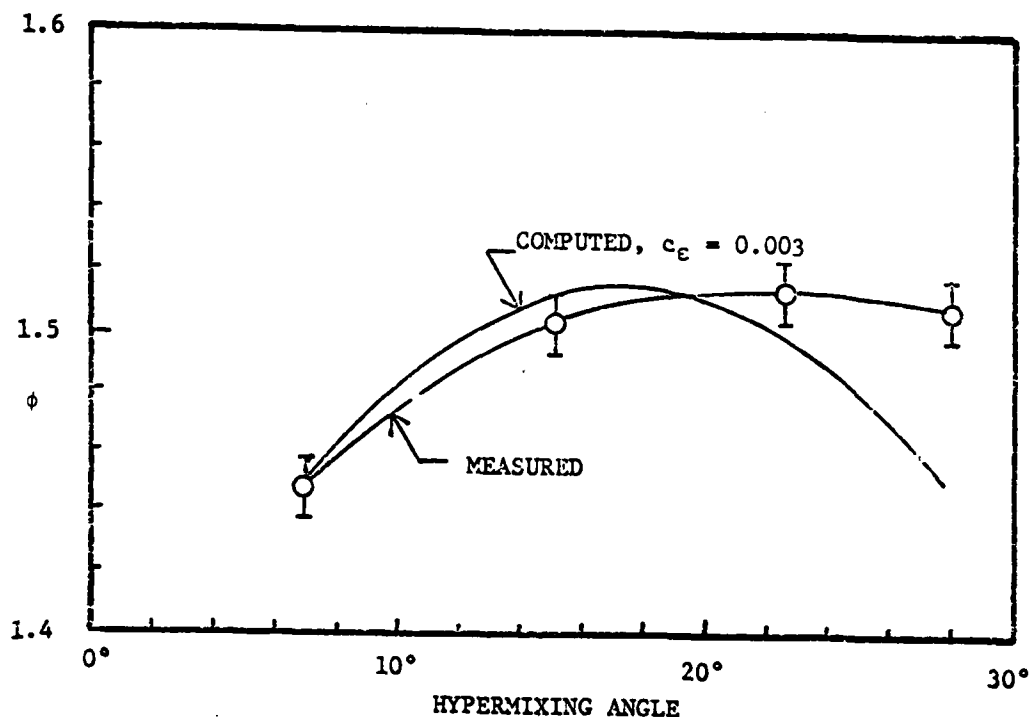
Figure 7-13 Comparison of Measured and Computed Augmentation

improve the calculations, it was decided to include the neglected swirl terms in the turbulence equations. In these equations, the term representing the generation rate of turbulent kinetic energy can be written in general form as:

$$G = -\rho \overline{u'_i u'_j} \frac{\partial U_i}{\partial x_j}$$

If the eddy viscosity model

$$-\overline{\rho u'_i u'_j} = \mu_t \left(\frac{\partial U_i}{\partial x_j} + \frac{\partial U_j}{\partial x_i} \right)$$

is used along with the cartesian coordinates notation, $U_1 = U$, $U_2 = V$, $U_3 = W$, $x_1 = x$, $x_2 = y$, and $x_3 = z$, then G is

$$G = \mu_t \left[\left(\frac{\partial U}{\partial y} \right)^2 + \left(\frac{\partial U}{\partial z} \right)^2 \right] + \mu_t \left(\frac{\partial W}{\partial y} + \frac{\partial V}{\partial z} \right)^2 + 2\mu_t \left[\left(\frac{\partial V}{\partial y} \right)^2 + \left(\frac{\partial W}{\partial z} \right)^2 \right]$$

If it is assumed that $U \gg V$, W and $\frac{\partial}{\partial y}$, $\frac{\partial}{\partial z} \gg \frac{\partial}{\partial x}$ the original form of the production term is obtained. The effect of including all the production terms is shown in Figure 7-14. The increase in augmentation is greatest at the largest angles, but the increase is not sufficient to significantly improve the predictions in this region. Thus, it was concluded that the applicability of the jet mixing analysis is limited to angles less than about 20°. Since the measured augmentation was found to have a maximum at small angles, this is not likely to be a serious limitation.
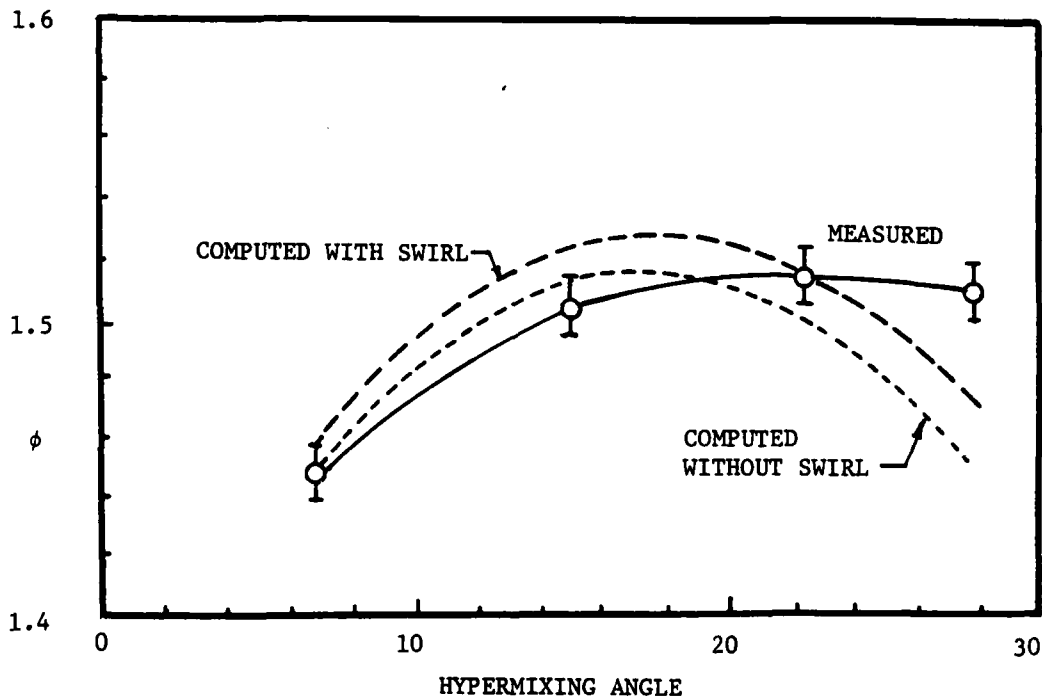
Figure 7-14 Effect of Swirl on Analytical Prediction

7.3   ASYMMETRIC NOZZLES

7.3.1   Introduction

The asymmetric cruciform centerbody nozzle design was developed in an
effort to capitalize on the generally superior entrainment characteristics
of the symmetric cross slot nozzle configuration while maintaining
packaging limits imposed by typical supersonic airfoil contours. This
design combines a series of aft facing spanwise slot nozzles with an
alternating series of cross slots on the forward side. The aft facing
span slots provide a flat surface contour for the airfoils upper surface
close-out while the cross slot is hidden in the airfoil cross section (see
Figure 7-15).

This basic nozzle design offers several packaging advantages over the
hypermixing nozzle concept. These advantages include elimination of an
upper surface close-out system and the removal of discontinuities from the
upper surface of the airfoil. In an effort to develop an asymmetric
centerbody nozzle, which would be compatible with the XFV-12A aircraft
design constraints while optimizing augmenter performance, a series of
asymmetric centerbody designs were evaluated.

Typically, during this series, nozzles were conceptually defined utilizing
both the available experimental data base and analytical trade studies.
Based on this information, if a performance gain was anticipated, the
nozzle was then designed and fabricated. Initial experimental evaluation
entailed measuring the isolated thrust efficiency (C_T) of the nozzle
design. Isolated nozzle efficiency (C_T) is an indication of the nozzles
isolated thrust performance and is defined as the ratio of the nozzles
measured thrust to its potential ideal thrust level. Utilizing the level
of C_T as a reference, it is possible to correct the measured
augmentation ratio for the observed efficiency differences between
nozzles. The value of this correction is that it points out the relative
gain or loss in mixing (entrainment) due to a nozzle configuration change
without biasing the nozzles performance for actual differences in internal
and external losses or differences in workmanship during fabrication.
This method allowed for direct comparisons with performance predictions
made utilizing the 3D TKE analytical technique which assumes that the
value of a nozzle thrust efficiency (C_T) is a constant.

Following the isolated performance evaluation, each centerbody nozzle was
then installed in the augmenter model for a series of performance
evaluations. Traversing pressure surveys were utilized to examine the
nozzles mixing characteristics and isolate the augmenters relative flow
quality.
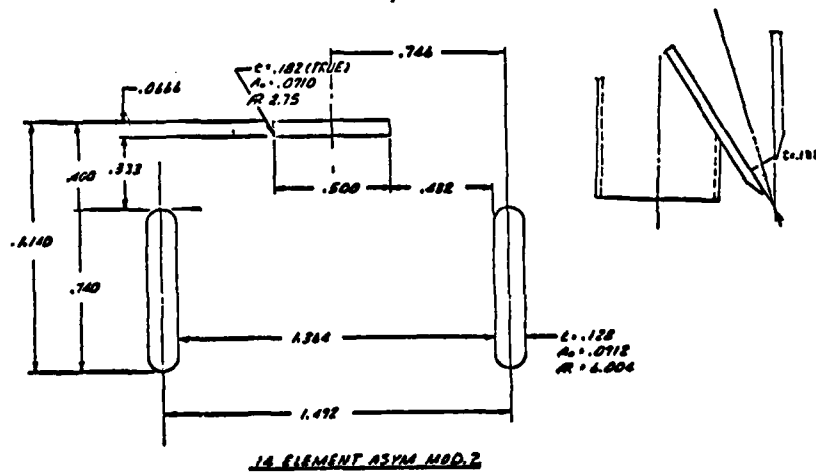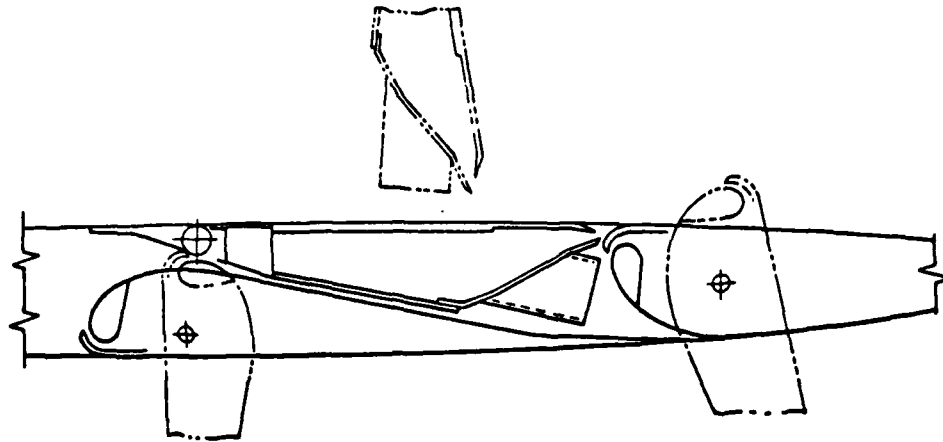
Figure 7-15   Asymmetric Centerbody Packaging Concept

The nozzles relative performance level was defined utilizing a mix of
analytical and experimental analysis techniques. Direct comparisons of
experimental data both isolated and in augmenter were employed. Measured
in augmenter performance levels were corrected for differences in isolated
nozzle thrust efficiency. This corrected data was compared to analytical
performance estimates to identify potential differences in a nozzles
entrainment characteristics. The remainder of this section will discuss a
number of the asymmetric nozzle designs along with pertinent results and
conclusions.

7.3.2   14 Element Asymmetric Centerbody Nozzle

The initial centerbody in this test series was designated as the 14
element asymmetric (see Figure 7-16). The basic design of this nozzle was
based on information which had been developed during previous augmenter
test programs; however, its specific geometric parameters were defined by
current aircraft packaging constraints. Utilizing the 3D TKE program,
analytical trade studies evaluating the performance of the baseline 14
element asymmetric centerbody compared to the 15° hypermixing centerbody
were conducted. While this analytical method had been used extensively to
evaluate hypermixing nozzle configurations, this was the first attempt to
evaluate asymmetric style nozzles. Based on available information
concerning the inlet conditions for this style nozzle, an increment of
$+.03 \Delta\phi$ was computed for the asymmetric design as compared to the 15°
hypermixing configuration. Subsequent to this initial computation, a
series of experimental inlet surveys were conducted on an asymmetric style
nozzle. The information was then utilized to update the inlet condition
modeling in the 3D TKE technique. Additional analytical trade studies
utilizing the improved inlet condition modeling indicated a performance
decrement of approximately $.03 \Delta\phi$ associated with the asymmetric style
nozzle.

Previous experimental trends indicated that asymmetric style nozzles were
capable of producing higher levels of performance than comparable
hypermixing nozzle designs. Based on this information, and in an effort
to optimize the performance potential of the asymmetric design, the nozzle
development effort continued. Isolated testing of the 14 element
asymmetric nozzle indicated that its nozzle efficiency (C_T) was
approximately $.03 \Delta C_T$ higher than the 15° hypermixing design (see
Figure 7-17).

Baseline in augmenter performance for this centerbody at the 3.24 inch
throat configuration, $A_2/A_0 = 15.2$, is presented in Figure 7-18.
Initially, the demonstrated performance was disappointing. However,
examination of the pressure survey data Test 280, Run 8 obtained during
this test series indicated that both the centerbody span slot and cross

30°

.830

2°

t = .128
A_0 = .0912
AR = 6.004

.746

.327

.710

1.364

1.492

t = .128 (TRUE)
A_0 = .0909
AR = 5.547

.353

.047

.400

.740

1.140
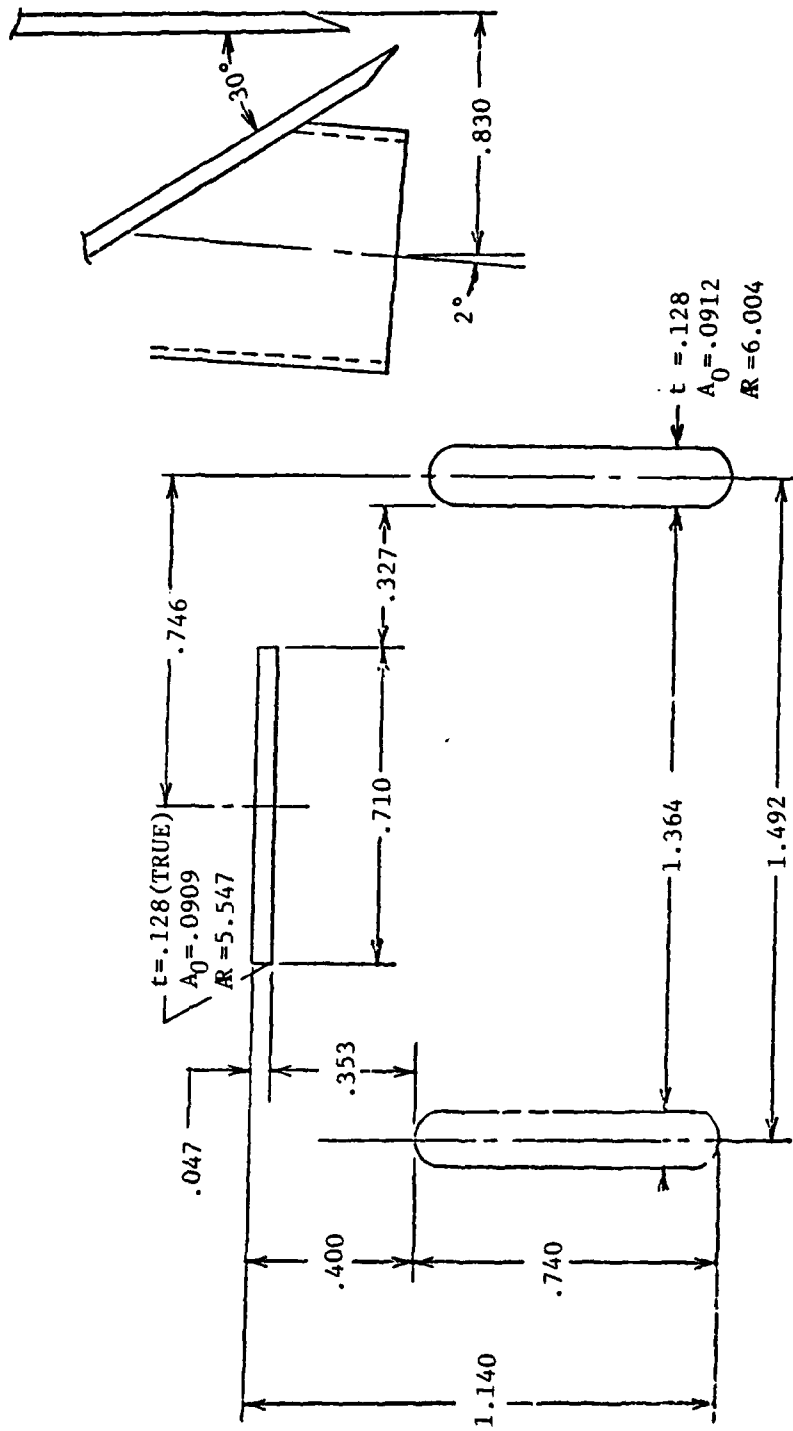
Figure 7-16 14 Element Asymmetric Nozzle
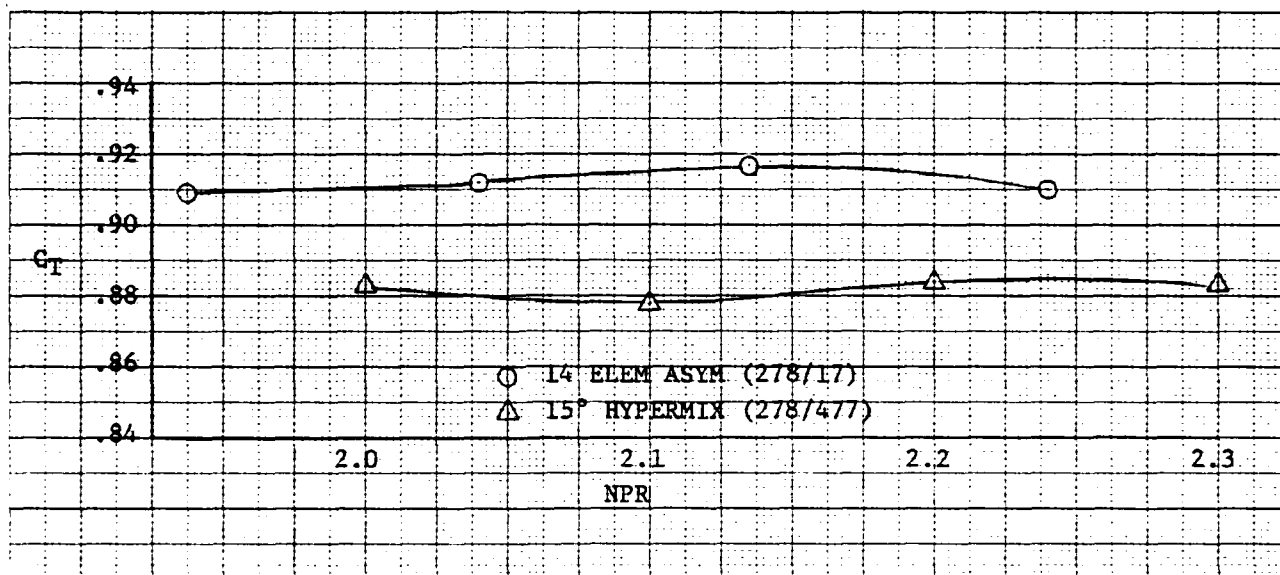
Figure 7-17   Isolated Centerbody Nozzle Efficiency (C_T)
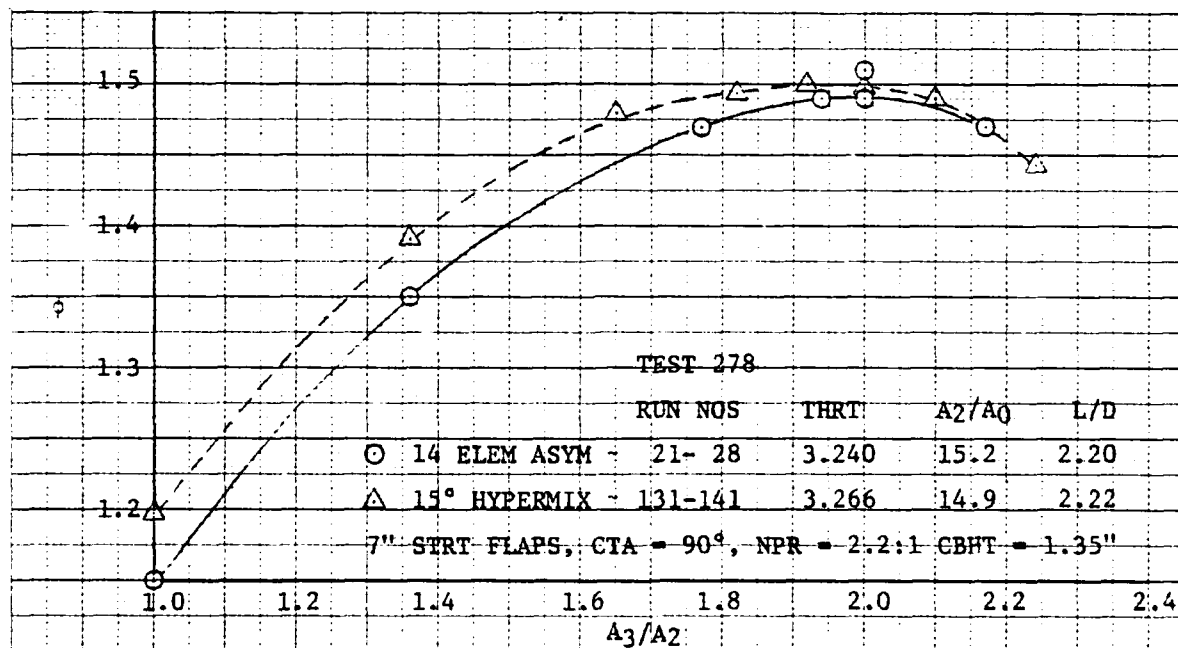
Figure 7-18   14 Element Asymmetric Versus 15° Hypermixing
Centerbody Nozzle A_3/A_2 Comparison

slot nozzles were strongly merged with the Coanda jets. Past experimental
experience indicated that this type of flow condition typically limited a
nozzles entrainment potential. Follow-on analytical trade studies were
conducted utilizing the 3D TKE program with inputs modified in a manner to
rotate the centerbody jet producing premature merging of the centerbody
and Coanda jets. The program indicated a large decrement in potential
performance could be expected from the highly merged jet flow condition.
Additional analytical studies indicated that enlarging the augmenter
throat from 3.24 inches to 4.42 inches (increasing the A_2/A_o from 15.2
to 20.2) would "unmerge" the nozzle jets and produce a performance
increment of +.06 $\Delta\phi$.

Figure 7-19 presents the experimental performance of the 14 element
asymmetric for a range of augmenter throat widths. The increment in
performance obtained by enlarging the throat was approximately +.04 $\Delta\phi$.
Pressure survey data indicated that the enlarged throat widths did
substantially reduce the merging of the centerbody-Coanda jets possibly
accounting for the increased levels of performance.

14 ELEM ASYM (UNMOD)

TEST 278

	RUN NOS	THRT	A_2/A_0	L/D
○	21-29	3.24	15.2	2.20
△	31-43	3.57	17.0	2.00
□	44-51	4.39	20.1	1.62

7" STRT FLAPS, CTA = 90°, NPR = 2.2:1, CBHT = 1.35, OLD COANDAS
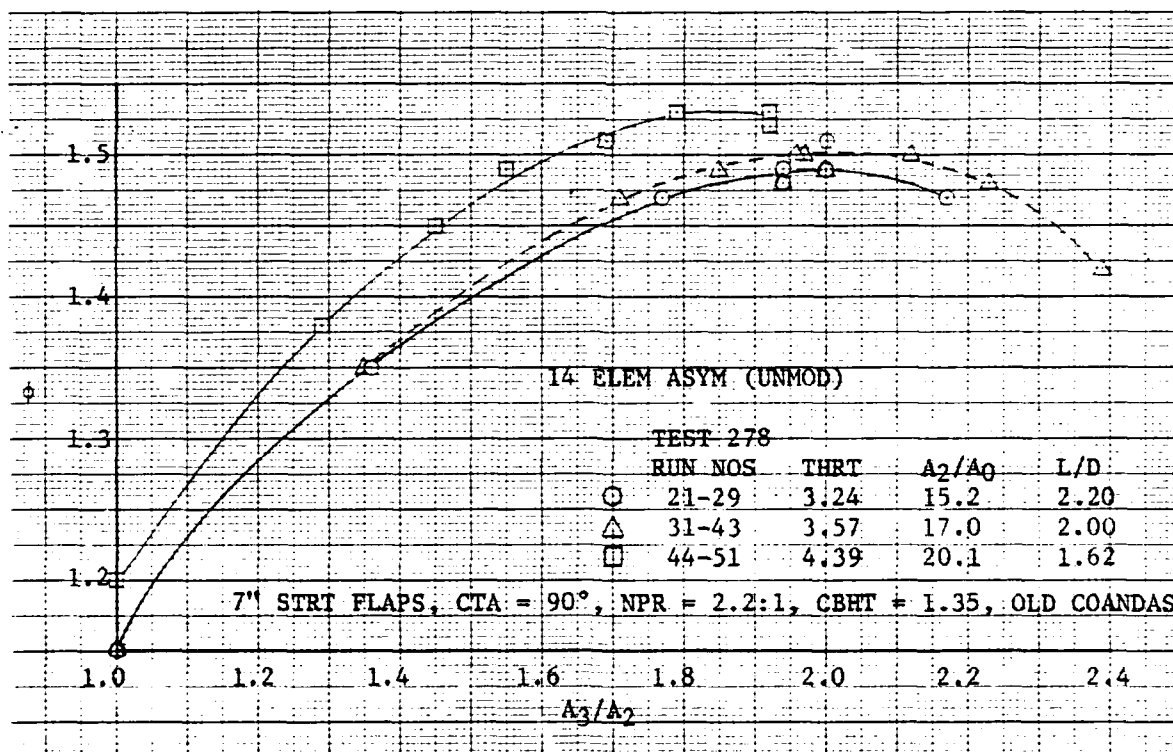
Figure 7-19 14 Element Asymmetric Throat Width Comparison

To determine if the centerbody and Coanda jets could be "unmerged" by
other means, a modification to three midspan cross slot elements was made
to reduce the relative launch angle between the two jets from 32° to
25.5°. A sketch of this nozzle configuration, designated 14 element
asymmetric mod 1, is presented in Figure 7-20. Survey data indicated that
the jets did unmerge, however, the region of $\Delta P_T = 0$ moved from between
the centerbody jets to the forward side between the diffuser jet and cross
slot jet. No improvement in entrainment or overall performance was noted
(see Figure 7-21).

7.3.3 14 Element Mod 2 Asymmetric Nozzle

In an attempt to improve the jet spreading (entrainment) characteristics
of the span slot nozzles while minimizing any adverse interaction between
the cross slot and span slot nozzles, the aspect ratio (AR) of these
nozzles was reduced from 5.55 to 2.75 (see Figures 7-16 and 7-22).
Analytical studies utilizing the 3D TKE program predicted a performance
increment of +.03 $\Delta\phi$ would be associated with the reduced AR design. This
increase was apparently the result of stronger span slot tip vortices
produced by the reduction of their aspect ratio.

Isolated centerbody nozzle performance tests indicated that the 14 element
mod 2 (reduced AR span slot) nozzle efficiency (C_T) was similar to the
14 element baseline asymmetric (see Figure 7-23). Apparently,
modifications necessary to reduce the span slot AR had little effect on
the isolated performance of the centerbody.

Figure 7-24 presents in-augmenter performance comparisons displaying a
performance increment of approximately +.03 $\Delta\phi$ demonstrated by the 14
element mod 2 (reduced span slot AR) nozzle. Comparisons of traversing
pressure surveys obtained at the augmenter exit indicated that the reduced
AR span slot did exhibit superior entrainment characteristics. This
improved entrainment was evidenced by increased jet spreading in the
chordwise Coanda jet direction while spanwise spreading characteristics
were similar.

The reduced AR span slot nozzles do, in fact, appear to increase the
entrainment characteristics of the asymmetric nozzle design without
notable reductions in isolated nozzle efficiency resulting in an overall
improvement in measured augmentation ratio.

7.3.4 14 Element Mod 3 Asymmetric Nozzle

It was hypothesized that two span slots in close proximity may produce a
set of co-rotating vortices resulting in increased levels of entrainment.
A further modification to this centerbody was made by replacing the single
span slots with a twin slot configuration as shown in the sketch of
Figure 7-25. This centerbody nozzle design was designated 14 element
asymmetric mod 3. Analytical trade studies utilizing the 3D TKE program
indicated that the twin span slot configuration could provide a modest
performance improvement (* .01 $\Delta\phi$) over its single span slot counterpart.

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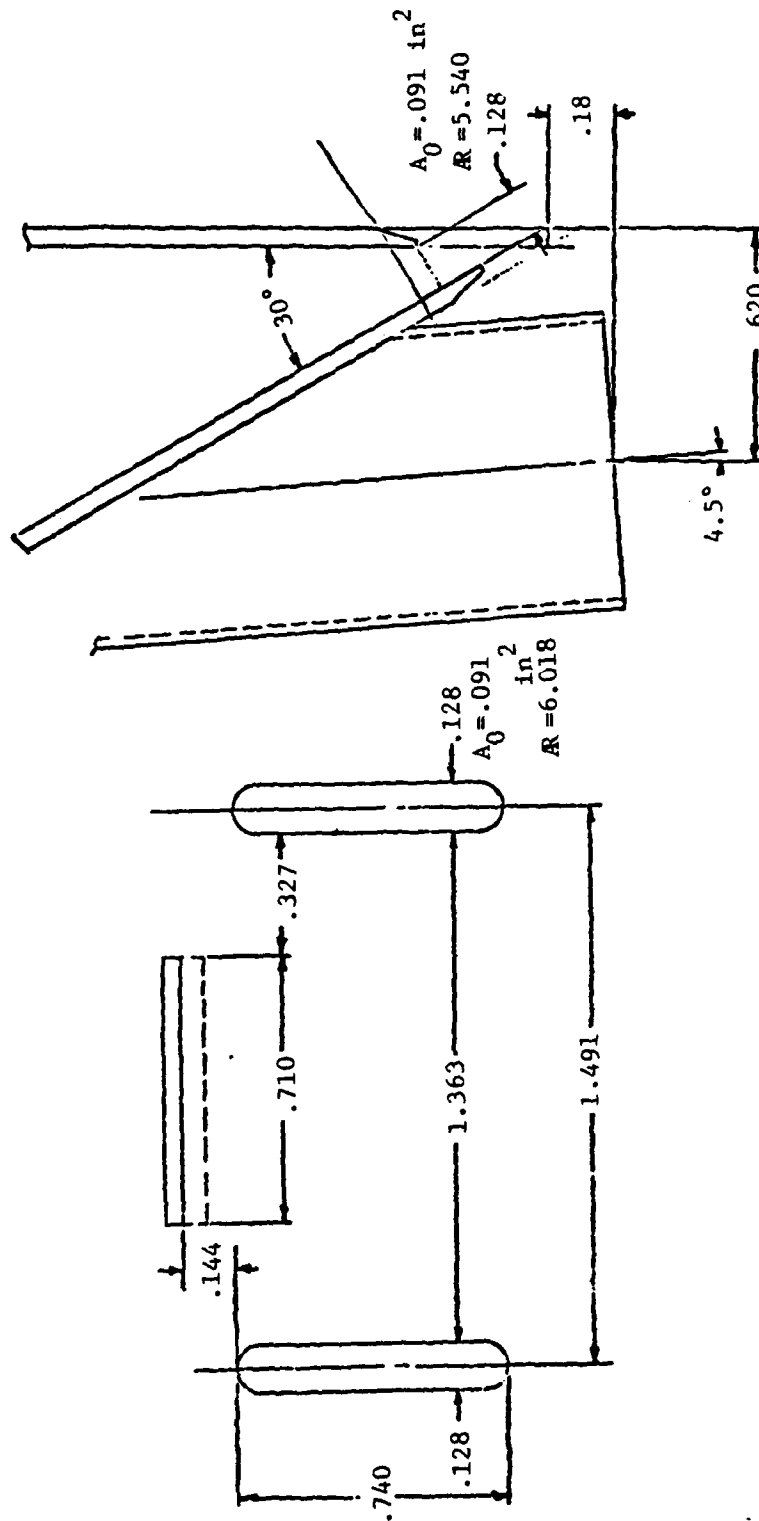
Figure 7-20 14 Element Asymmetric Mod No. 1
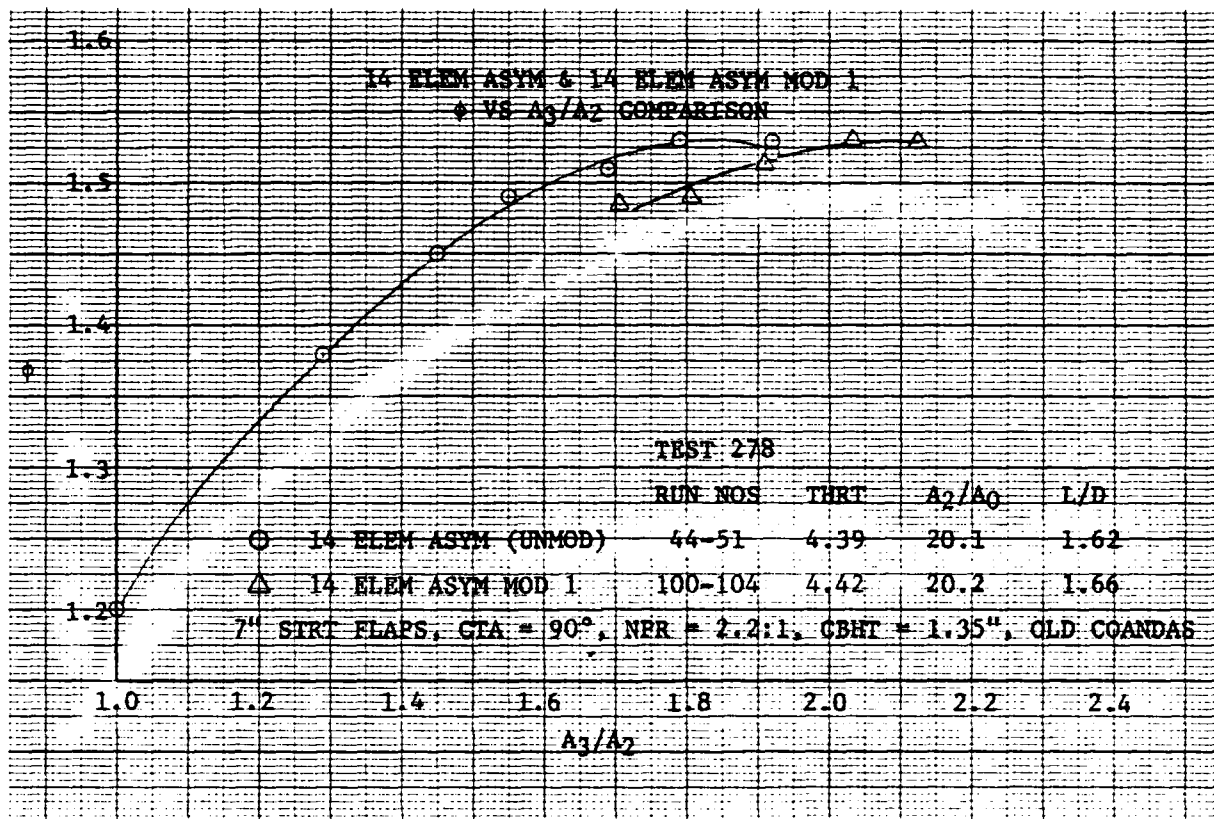
Figure 7-21   14 Element Asymmetric and 14 Element Asymmetric Mod No. 1
Comparison

t=.182

t=.128
$A_0 = .0912$
$R = 6.004$

.746

.432

.500

$t = .182 (TRUE)$
$A_0 = .0910$
$R = 2.75$

1.364

1.492

.0666

.333

.400

.740

1.140
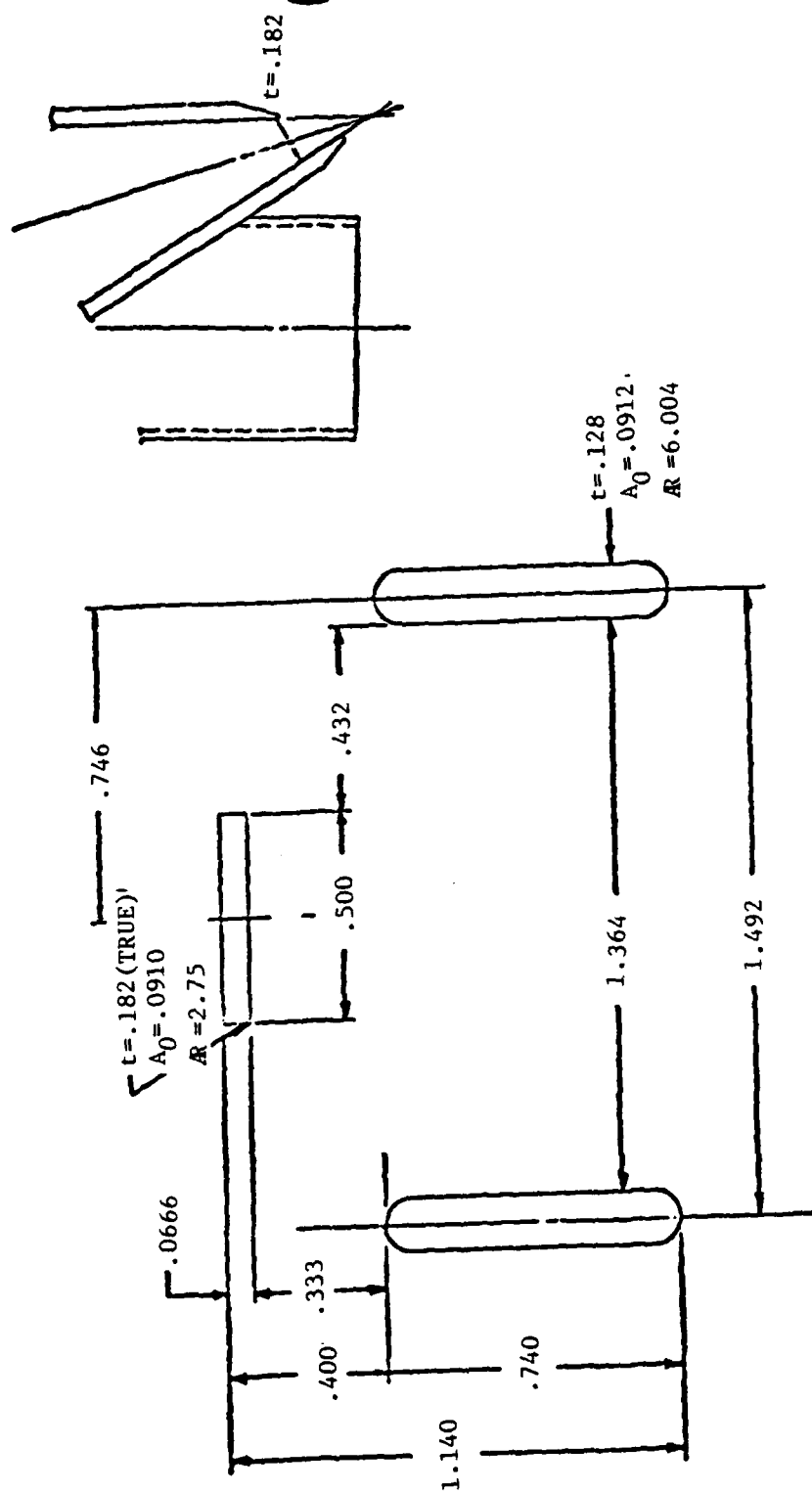
Figure 7-22 14 Element Asymmetric Mod No. 2
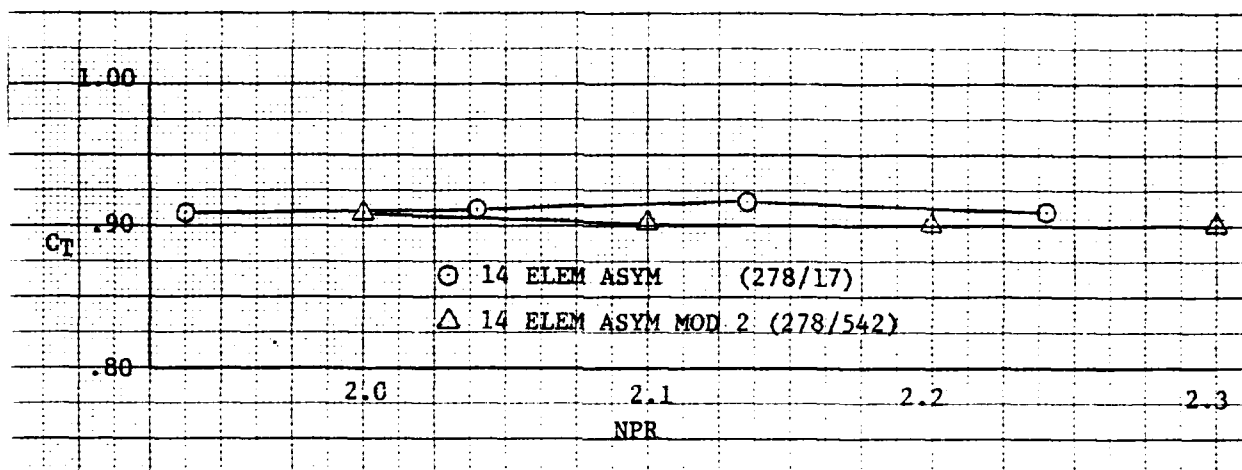
Figure 7-23   Isolated Nozzle Efficiency

Legend on chart:
⊙ 14 ELEM ASYM   (278/17)
△ 14 ELEM ASYM MOD 2 (278/542)

TEST 278				
	RUN NOS	THRT	A_2/A_0	L/D
⊙ 14 ELEM ASYM	44-51	4.390	20.10	1.62
△ 14 ELEM ASYM MOD 2	303-313	4.421	19.71	1.66

7" STRT FLAPS, CTA = 90°, NPR = 2.2:1, CBHT = 1.35"
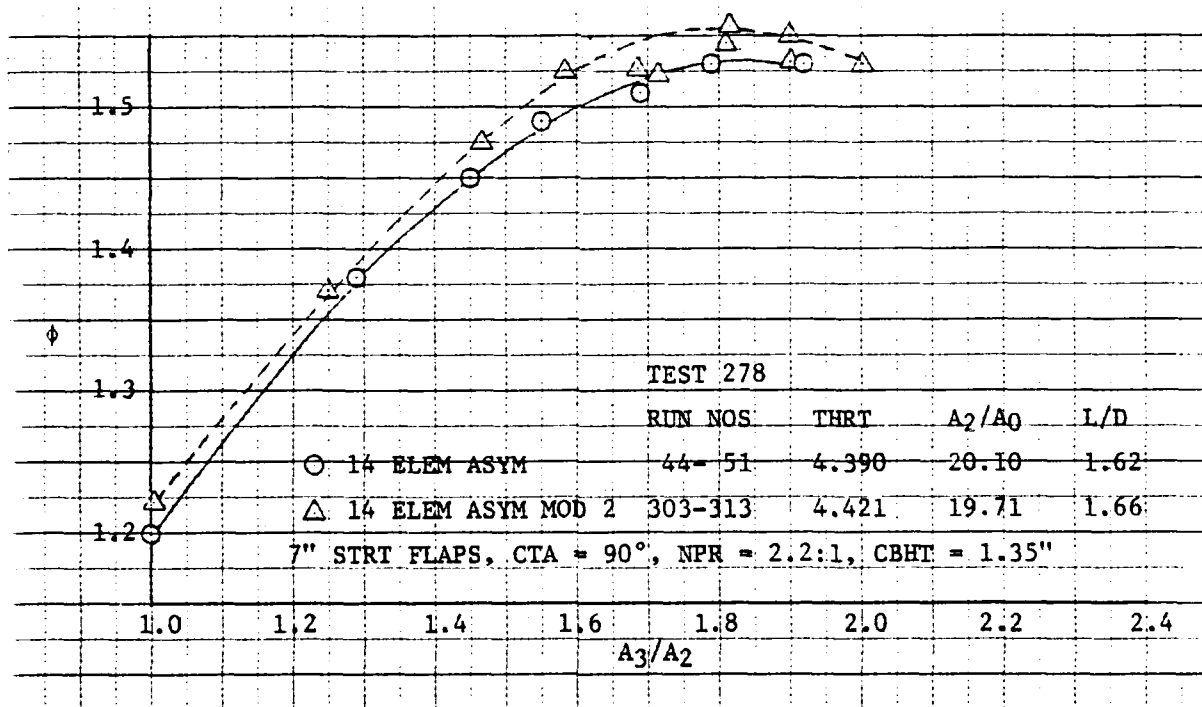
Figure 7-24   14 Element Asymmetric and 14 Element Asymmetric Mod No. 2
Comparison

(DWG. #356-01052-11)

t=.129

TOTAL A_0 = 2.6459 IN.^{2}

t=.129
A_0=.046
R=2.75

t=.129
A_0=.0912
R=6.004

.142

.354

.372

.354

.142

.047

.353

.400

.740

.128

1.364

1.492

1.140
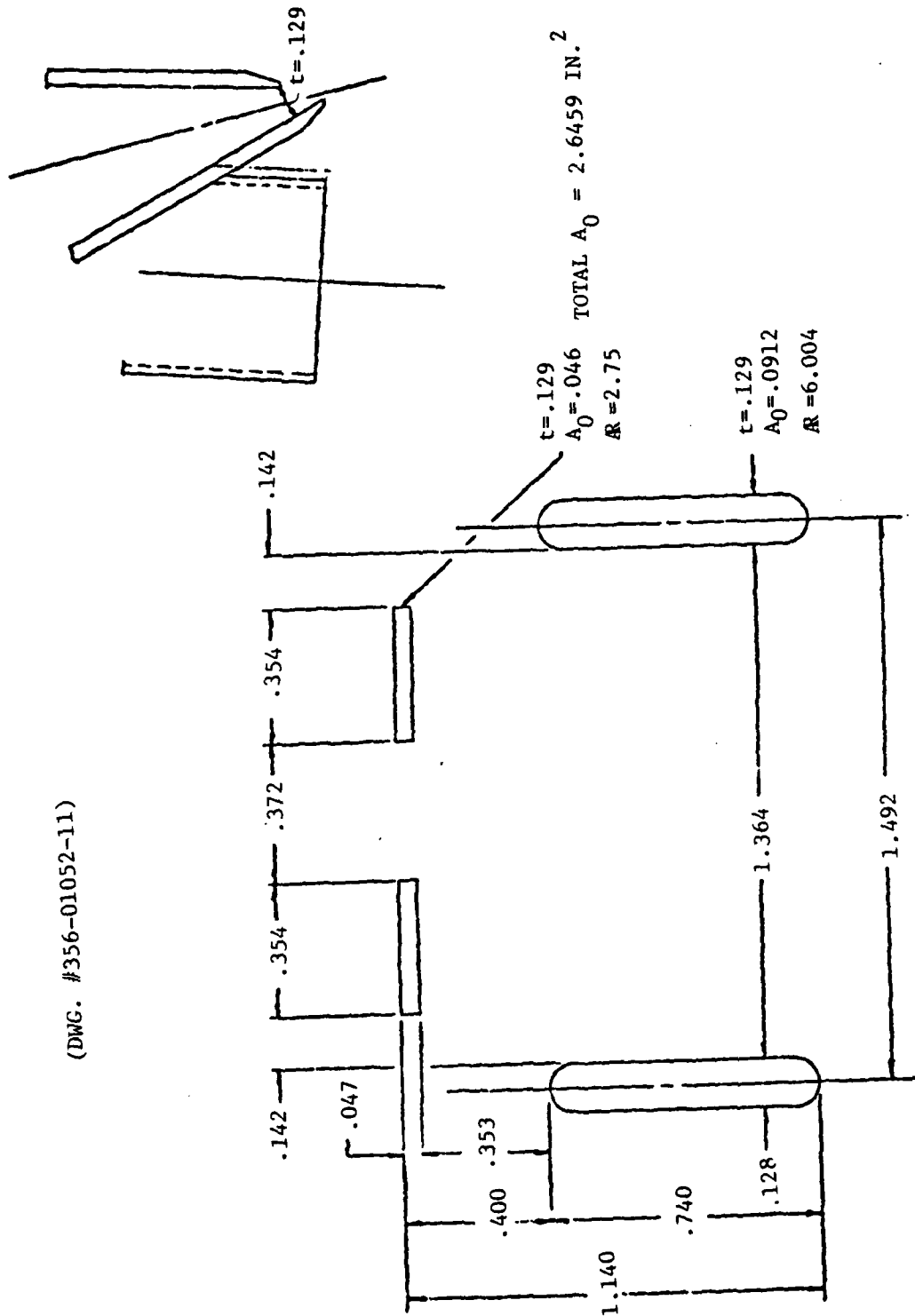
Figure 7-25   14 Element Asymmetric Mod No. 3

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Isolated performance investigations of the 14 element mod 3 (twin span slot) design indicated that this nozzle's thrust efficiency, C_T, was significantly lower than the mod 2 (single span slot) configuration (see Figure 7-26). The in-augmenter performance of the mod 3 design is presented in Figure 7-27. While the nozzle's overall measured performance is about .025 $\Delta\phi$ less than the mod 2 design, it should be noted that this performance deficiency can easily be attributed to the mod 3 nozzle's isolated performance. The measured decrement in C_T for the mod 3 nozzle would account for approximately - .05 $\Delta\phi$ decrement in total augmenter performance assuming this nozzle's entrainment was comparable to the mod 2 (single span slot) design. Apparently, the increased levels of entrainment produced by the twin span slot design offset a large portion of the C_T decrement resulting in the measured penalty in augmentation compared to its single span slot counterpart. The decrement in isolated performance, C_T, may be attributable to increased skin friction drag produced by the twin span slots additional internal nozzle perimeter or may be, in part, due to flaws in workmanship resulting from nozzle modification.

It appears that while the twin span slot design may produce additional levels of mixing (entrainment) as indicated by the 3D TKE analysis, these increased levels of mixing are inadequate to offset the large penalty in nozzle efficiency associated with this design.
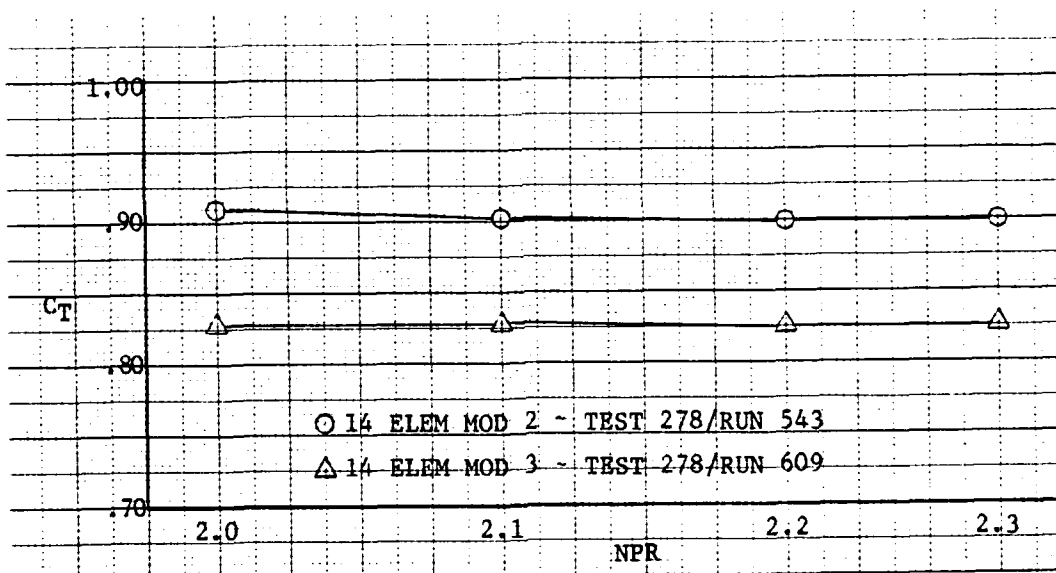
Figure 7-26   Isolated Nozzle Efficiency (C_T)

14 ELEM ASYM MOD 2 & MOD 3
φ VS A3/A2 COMPARISON

TEST 278

	RUN NOS	THRT	A2/A0	L/D
⊙ 14 ELEM MOD 2	579-587	5.031	21.70	1.49
△ 14 ELEM MOD 3	621-627	5.028	21.65	1.49

7" STRT FLAPS, CTA = 90°, NPR = 2.2:1, CBHT = 1.60"
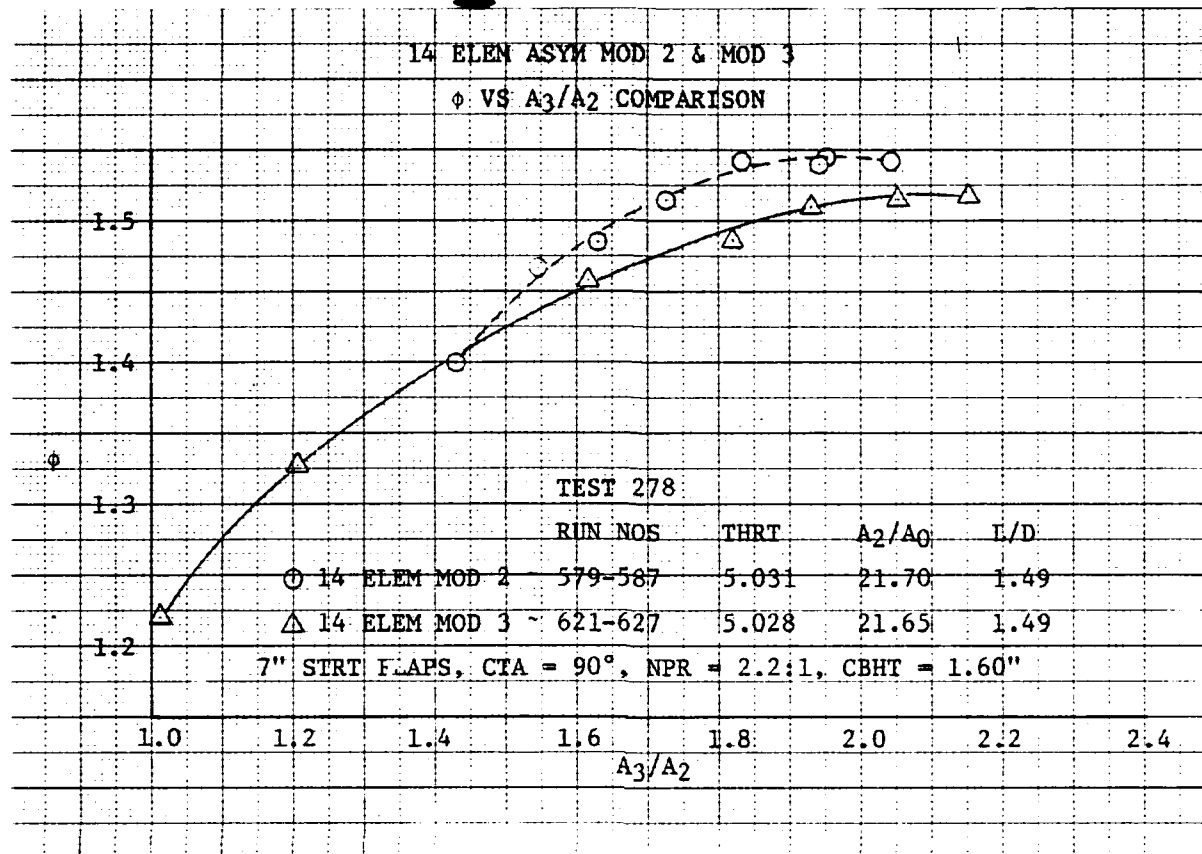
Figure 7-27 14 Element Asymmetric Mod 2 and Mod 3 Comparison

7.3.5   18 Element Asymmetric

In an effort to determine if the number of cross slot elements had an effect on centerbody performance, an 18 element asymmetric centerbody nozzle was designed and fabricated (see Figure 7-28). Geometrically, this nozzle was similar in design to the 14 element mod 2 asymmetric. That is, a series of race track cross slots alternating in combination with a series of single span slot nozzles. Exit area flow split and relative nozzle launch angles were all similar to the 14 element mod 2 centerbody configuration.

Analytical studies utilizing the 3D TKE technique indicated a performance increment of approximately +.04 Δφ could be expected from increasing the number of cross slot nozzle from 14 to 18. Again, during the analytical trade study, isolated nozzle thrust efficiency was assumed to be the same for both the 14 element and 18 element designs. Apparently, the predicted performance increase was the result of improved mixing (entrainment) characteristics due to the increased number of cross slot nozzles.

Figure 7-29 presents a comparison of the measured values of nozzle efficiency (C_T) for the 14 element and 18 element designs. The decrement of approximately -.04 to .05 ΔC_T can be attributed to the
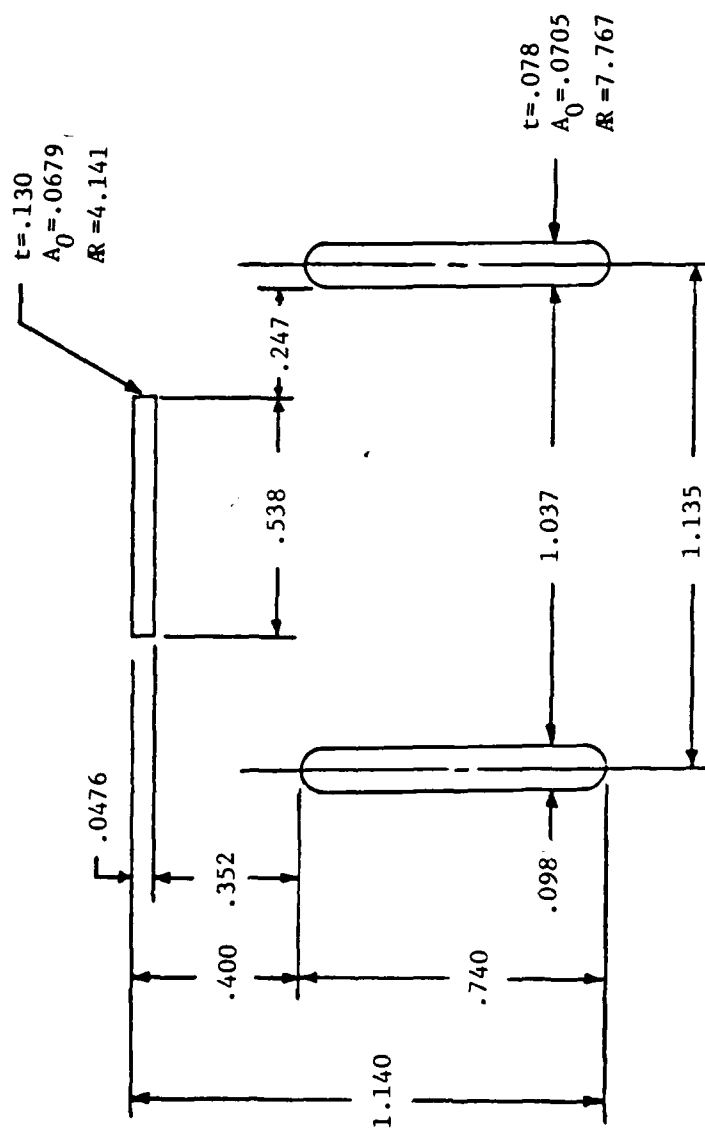
Figure 7-28 18 Element Asymmetric Nozzle

$t=.130$
$A_0=.0679$
$R=4.141$

$t=.078$
$A_0=.0705$
$R=7.767$

.130

.247

.538

1.037

1.135

.0476

.352

.400

.740

.098

1.140
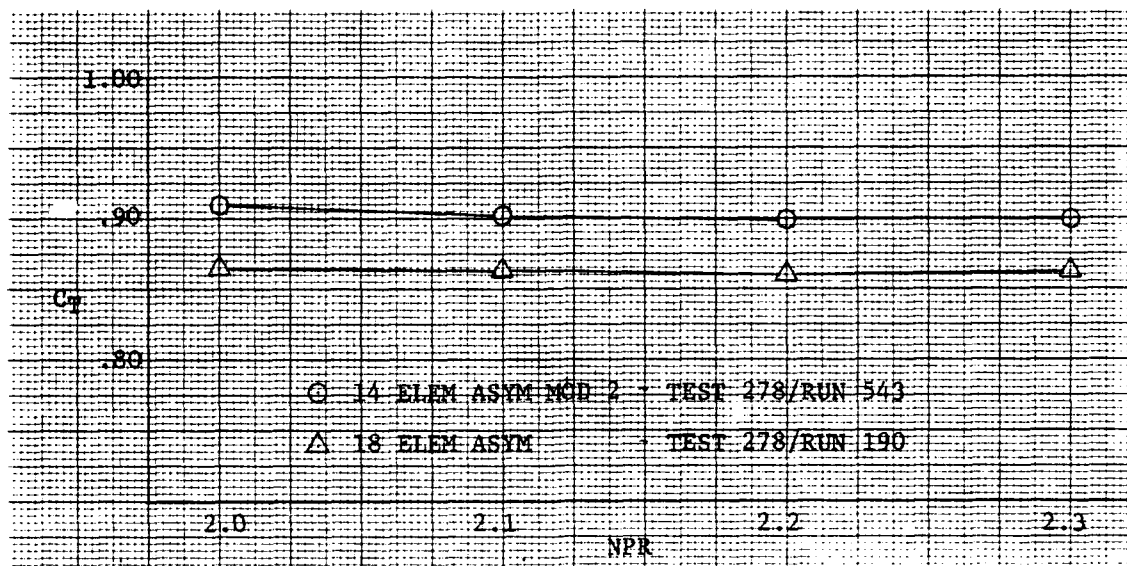
Figure 7-29   14 Element Mod 2 and 18 Element Asymmetric Comparison
Isolated Nozzle Efficiency (C_T)

increased nozzle perimeter resulting in additional losses in nozzle
efficiency due to increased skin friction and nozzle boundary layer
growth. A comparison of in-augmenter performance of ϕ versus A_3/A_2
presented in Figure 7-30 shows a loss in performance of $-.04 \, \Delta\phi$ for the 18
element design compared to its 14 element counterpart. Correcting the 18
element's augmentation deficiency for its measured decrement in isolated
nozzle efficiency accounts for approximately .045 of the $\Delta\phi$ loss. The
loss in nozzle efficiency due to the addition of cross slot elements
appears to offset any potential improvement in mixing entrainment produced
by the additional cross slot elements.

7.4   SYMMETRIC CENTERBODY NOZZLE

7.4.1   Introduction

An improvement in augmentation over hypermixing of about .04 was obtained
with the initial tests of the asymmetric nozzles. Further improvements
may have been obtained through a test/analysis optimization program.
However, it was uncertain how long such a program would take or how much
improvement could be obtained. Therefore, it was decided to
experimentally determine the maximum that could be gained by utilizing
full width cross slot nozzles. In Reference (26) such a nozzle was shown
to produce performance gains.

The graph shows ϕ on the y-axis (ranging 1.4 to 1.6) versus A_3/A_2 on the x-axis (1.0 to 2.2).

		TEST 278			
		RUN NOS	THRT	A_2/A_0	L/D
○	14 ELEM MOD 2	315-320	4.421	19.71	1.66
△	18 ELEM ASYM	225-230	4.420	19.53	1.66
	7" STRT FLAPS, CTA = 90°, NPR = 2.2:1, CBHT = 1.60"				
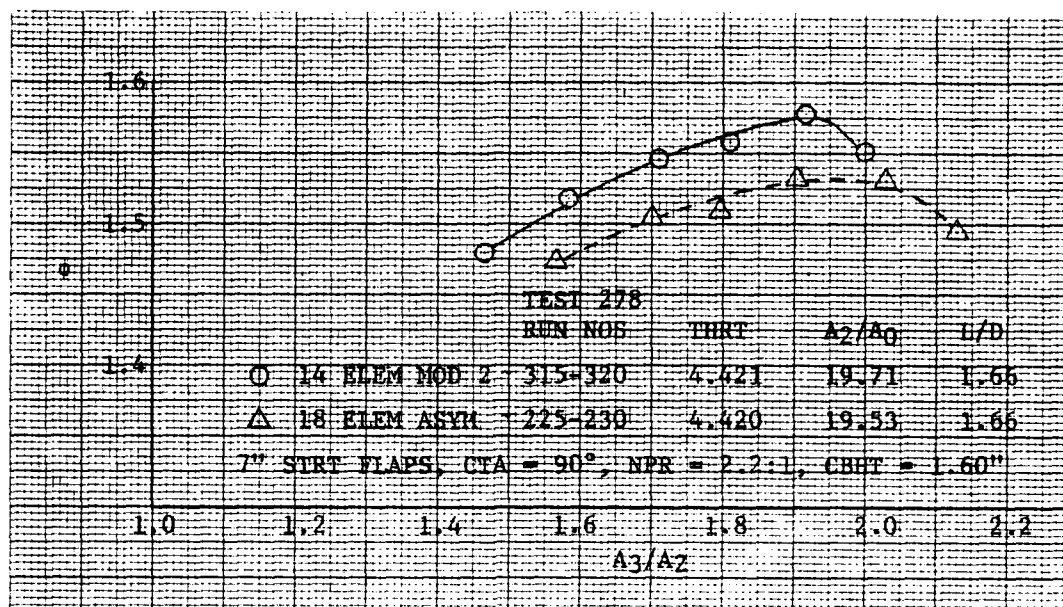
Figure 7-30   14 Element Mod 2 Asymmetric and 18 Element Asymmetric
ϕ Versus A_3/A_2 Comparison

The nozzle used in Reference (26) was adapted for testing in the XFV-12A
wing augmenter. A sketch of the nozzle is shown in Figure 7-31 and the
results of the test are presented in Figure 7-32. This result was very
encouraging and indicated a potential for the symmetric nozzles. However,
the BT-3 nozzle was too wide for packaging within the XFV-12A. Therefore,
a reduced width nozzle was design, fabricated, and tested. A sketch of
this nozzle is shown in Figure 7-33 and the test results shown in Figure
7-32 were very disappointing. A decrement in augmentation ratio of
approximately .07 was obtained.

In order to improve the performance of the reduced width nozzle and yet
maintain a packageable design, a program was initiated to develop a high
performance symmetric cross slot nozzle.

The nozzle development program consisted of a systematic series of tests
to determine ejector performance and flow characteristics as a function of
nozzle geometric parameters. The length of time required between nozzle
design and test necessitated defining a matrix of nozzles designed to
study cross slot width, span slot and cross slot aspect ratios, span
slot/cross slot flow split, bowtie ratio and initial jet flow angularity.
A parallel study was conducted in which the jet mixing analysis computer
code was utilized. Correlation between test and analysis would lead to a

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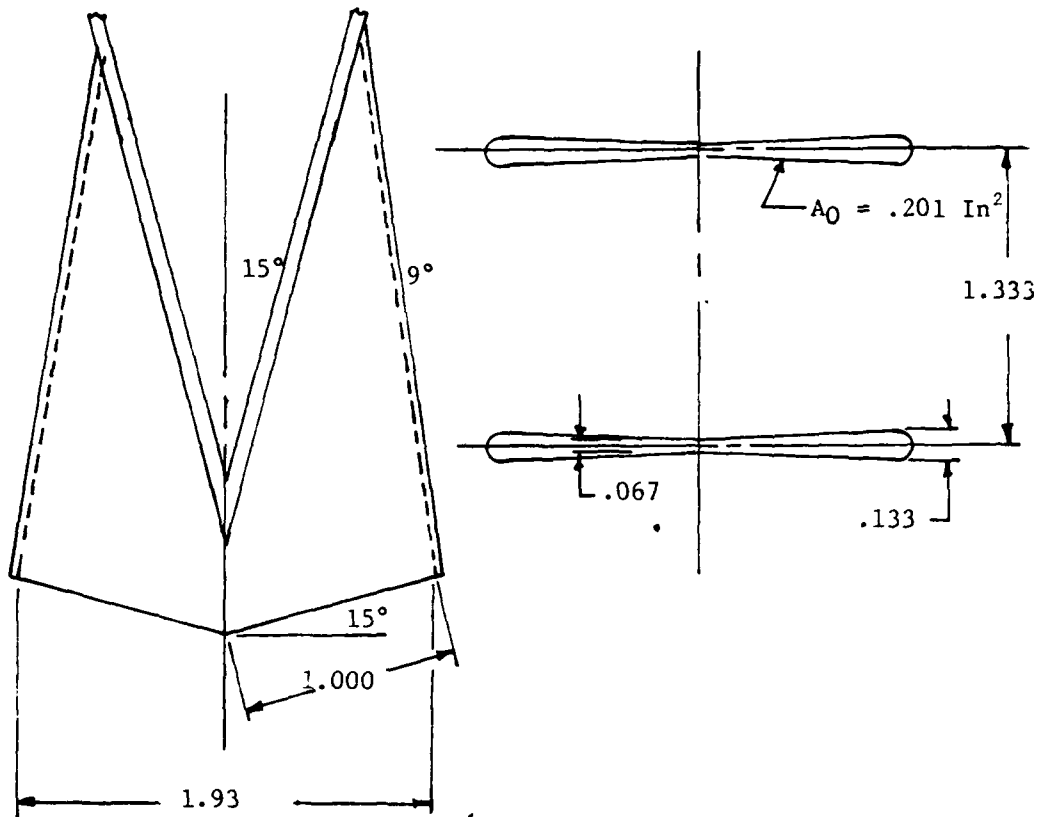
Figure 7-31 Symmetric Cross-Slot Nozzle (BT-3)

better understanding of the complex interactions between nozzle geometry and ejector flow characteristics along with the capability to define a packageable high performance symmetric cross slot nozzle.

7.4.2   Bowtie Ratio Effects

The cross-slot nozzle bowtie ratio (BTR) is defined as the ratio of the nozzle thickness at the tip to the nozzle thickness at the centerline. The purpose of increasing BTR is to increase entrainment by placing more primary flow into the vortex structure at the cross-slot tip.

The effect of increased BTR on ejector performance was obtained both analytically and experimentally. Experiments were conducted on centerbody nozzles with BTR's of 1, 2, and 3. All nozzle and ejector parameters were held constant except BTR.

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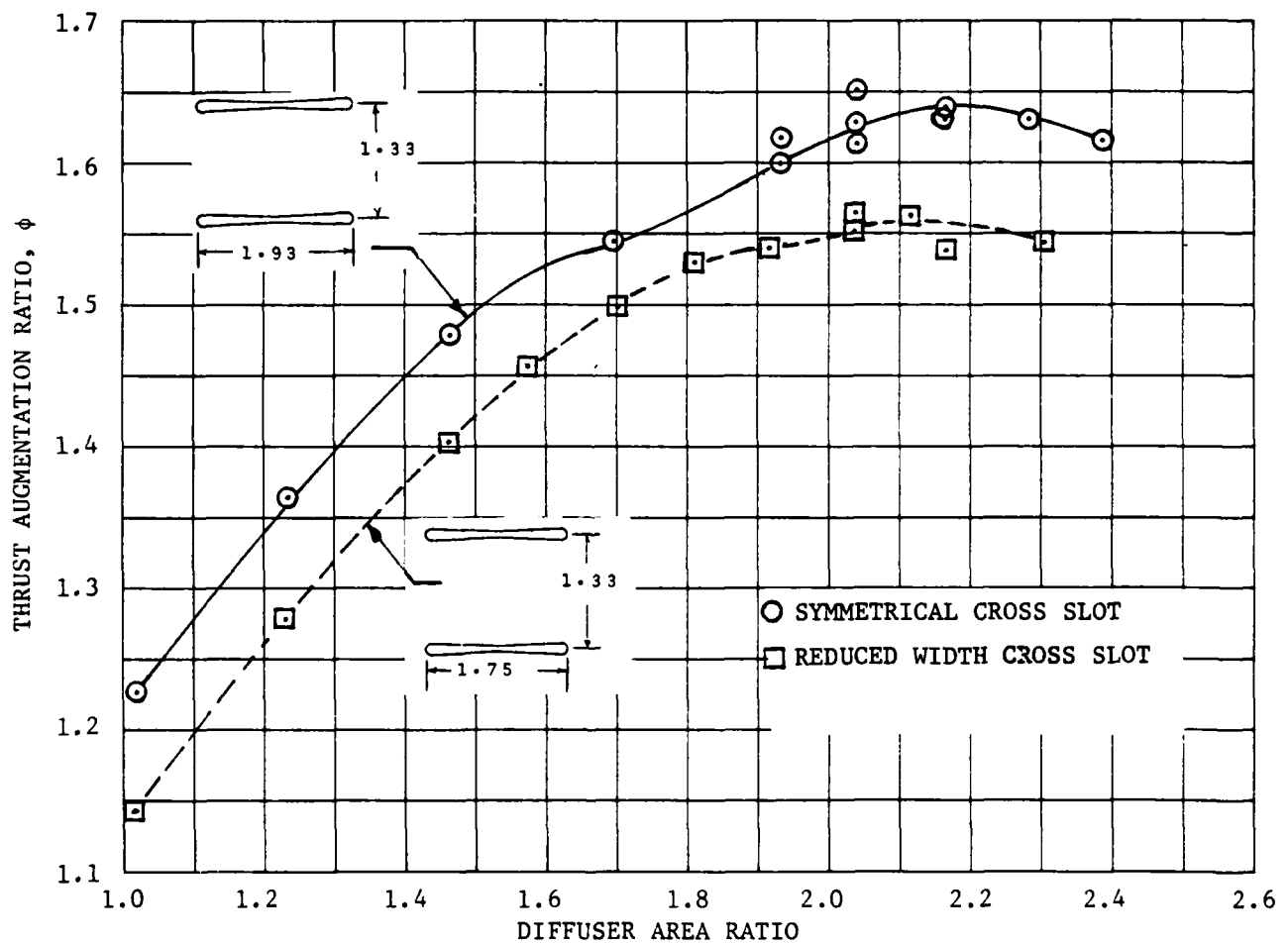
Figure 7-32 Symmetric Cross-Slot Nozzle Thrust Augmentation Versus
Diffuser Area Ratio

Rockwell International
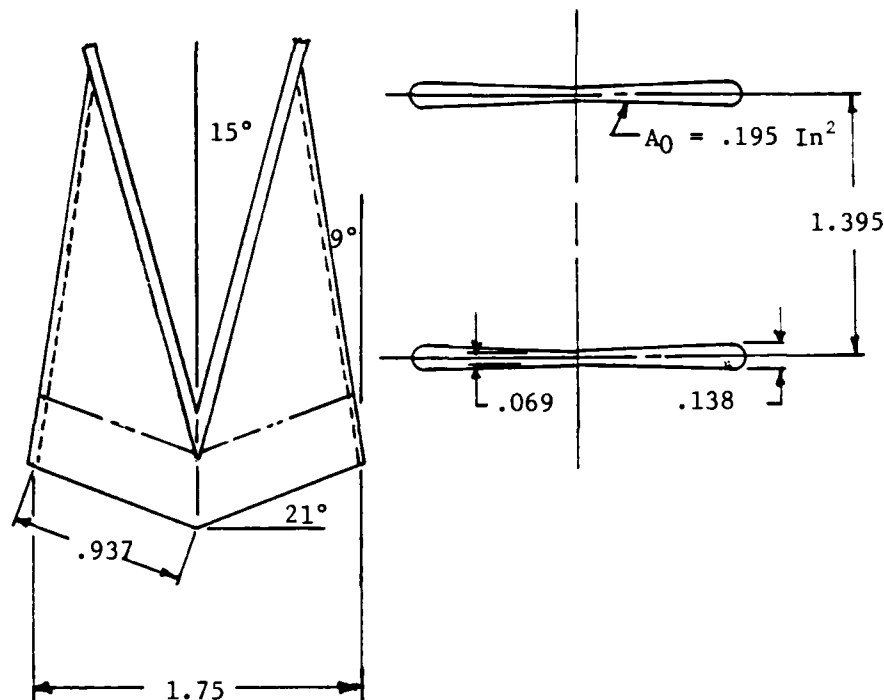
Figure 7-33 Reduced Width Symmetric Cross Slot Nozzle

Analytically, a linear jet angle distribution was assumed for the cross-slot nozzle and was the same for all BTR's. Figure 7-34 presents a typical comparison of the measured and assumed distributions for one nozzle lobe. The nozzle thrust coefficient was assumed to be 0.925.

A comparison of the experimental and analytical ejector performance results are shown in Figure 7-35 as $\Delta\phi$ versus BTR for a pressure ratio = 2.2 and a diffuser area ratio of 2.0. The experimental results have been adjusted for C_T effects i.e., corrected by the experimentally determined C_T's shown in Figure 7-36.

The analytical development of the jets for BTR = 1, 2, and 3 are shown in Figures 7-37, 7-38, and 7-39, respectively. The axial velocity profiles at three streamwise stations corresponding to the ejector throat, a point approximately midway through the diffuser and the ejector exit, are shown on the left. The convection velocities in the transverse planes at the throat and midway through the diffuser are shown on the right. The exit plane is not shown because the transverse velocities have virtually decayed by that station. The spanwise axis runs from left to right along the base of each profile, while the chordwise axis runs from bottom to top along the side. Note that the spanwise scale on the axial plots have been elongated to show detail. In the transverse planes, each velocity vector is centered on a grid point; the surface of the axial velocity profiles is

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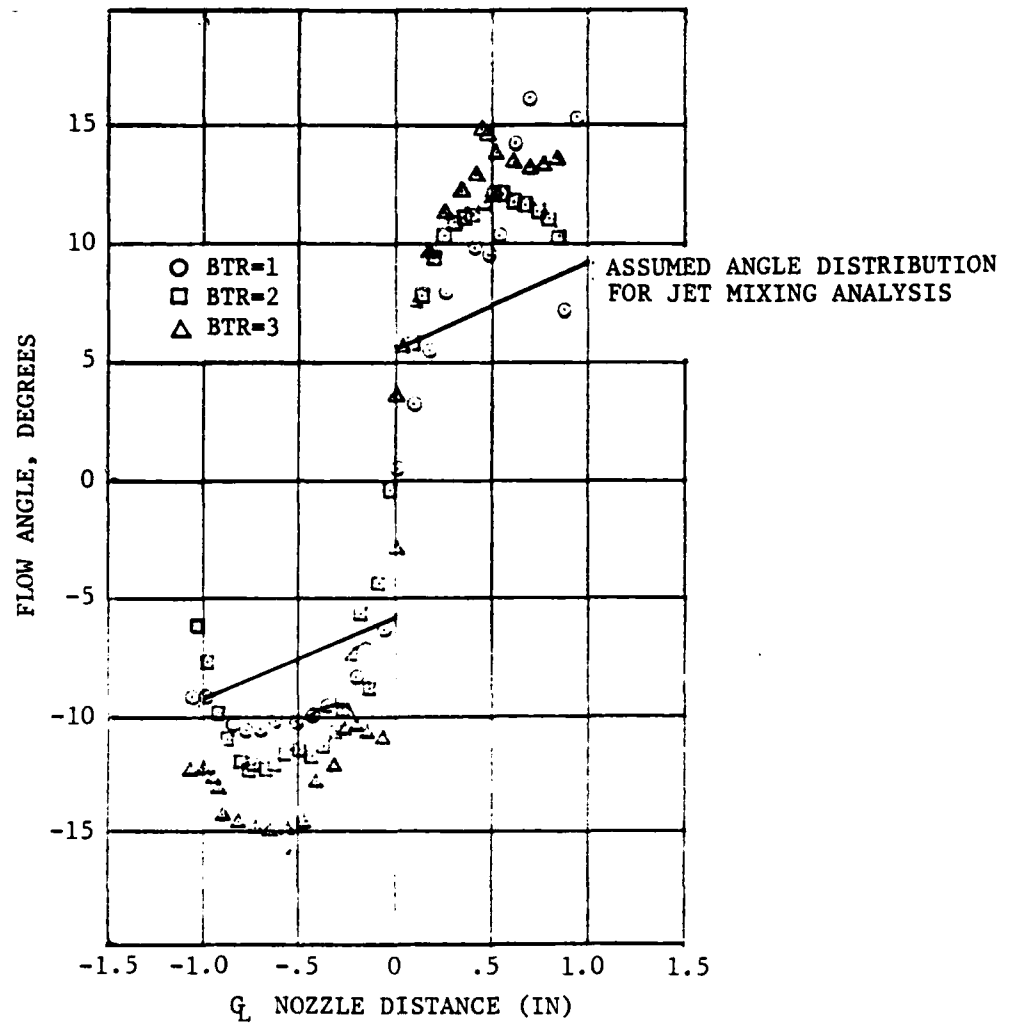
Figure 7-34 Variation of Jet Flow Angle for Various Nozzle Bowtie Ratios
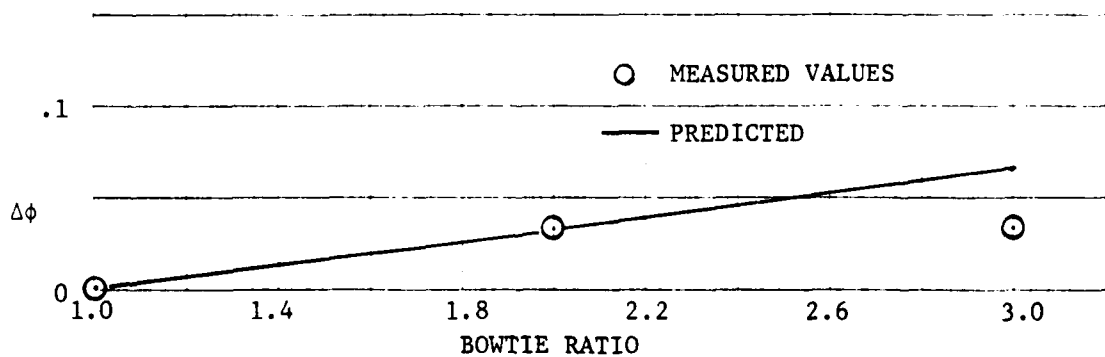
Figure 7-35 Effect of Bowtie Ratio On Thrust Augmentation

Rockwell International
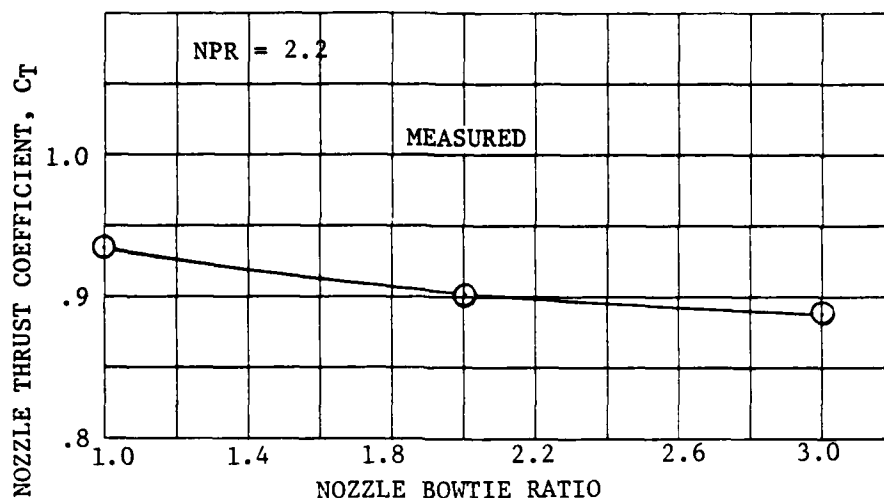
Figure 7-36 Effect of Nozzle Bowtie Ratio on Nozzle Thrust Coefficient

defined by lines passing over these points. The near Coanda jet in the axial velocity profiles has been omitted for clarity. The left hand symmetry plane runs through the center of a span slot and the right hand symmetry plane runs through the cénter of a cross slot.

One side of the vortex pair which develops at the ends of each cross slot are visible in the transverse planes. The mixing action of these vortices increases the entrainment in this region, which causes the jet to develop the "dog bone" shaped cross section shown schematically in Figure 7-40. The influence of the increased BTR is seen in the transverse plane at the throat and by the increased spanwise spreading of the cross slot at the exit.

The experimentally obtained axial velocity plots at the throat are shown in Figure 7-41 and the exit axial velocity plots are shown in Figure 7-42.

It is concluded that increases in BTR result in enhancement of the "Dog Bone" vortices and hence increased entrainment and augmentation. However, the experimental results showed a degradation in nozzle thrust coefficient with increased BTR, which negated the benefits of increased mixing. However, it may be possible to design an increased BTR nozzle with a high C_T.

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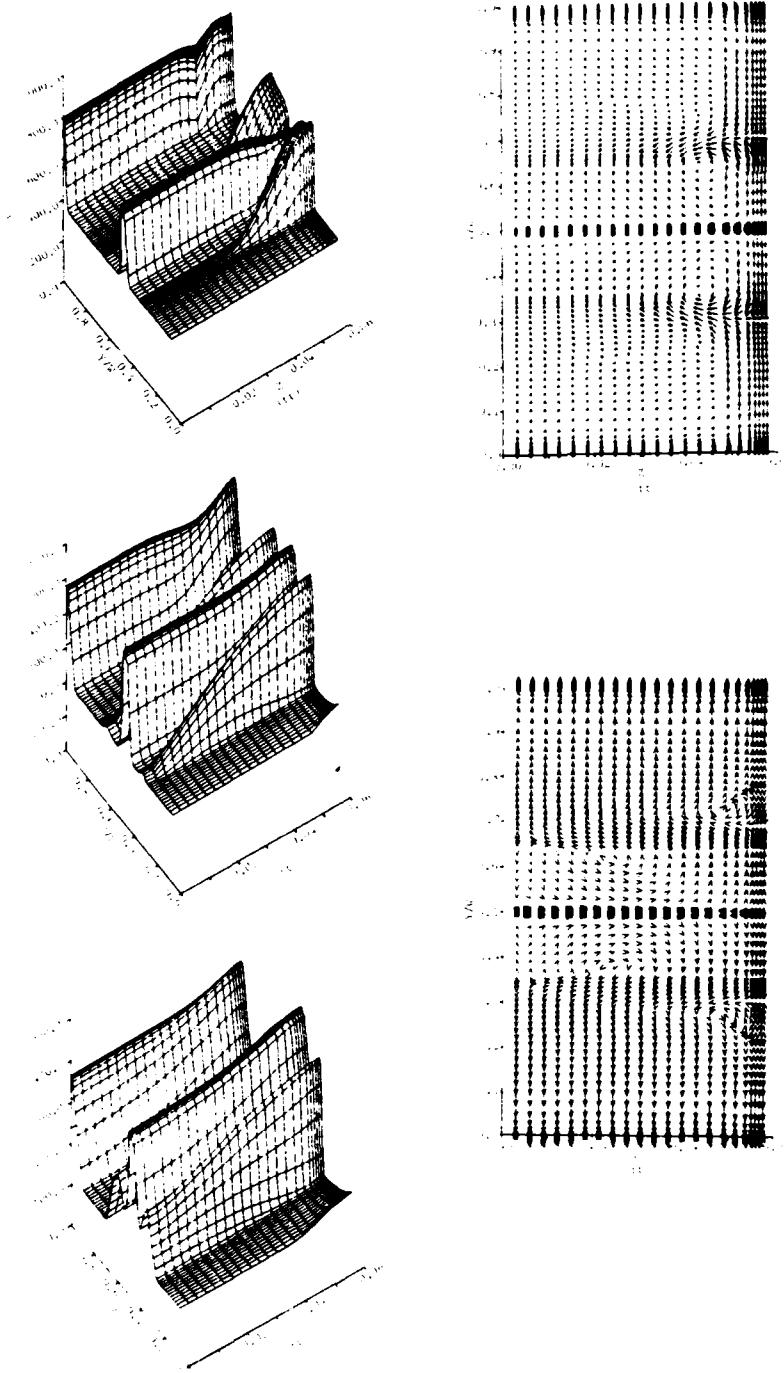
Figure 7-37   Bowtie Ratio 1 Nozzle
(Study Baseline)

Rockwell International
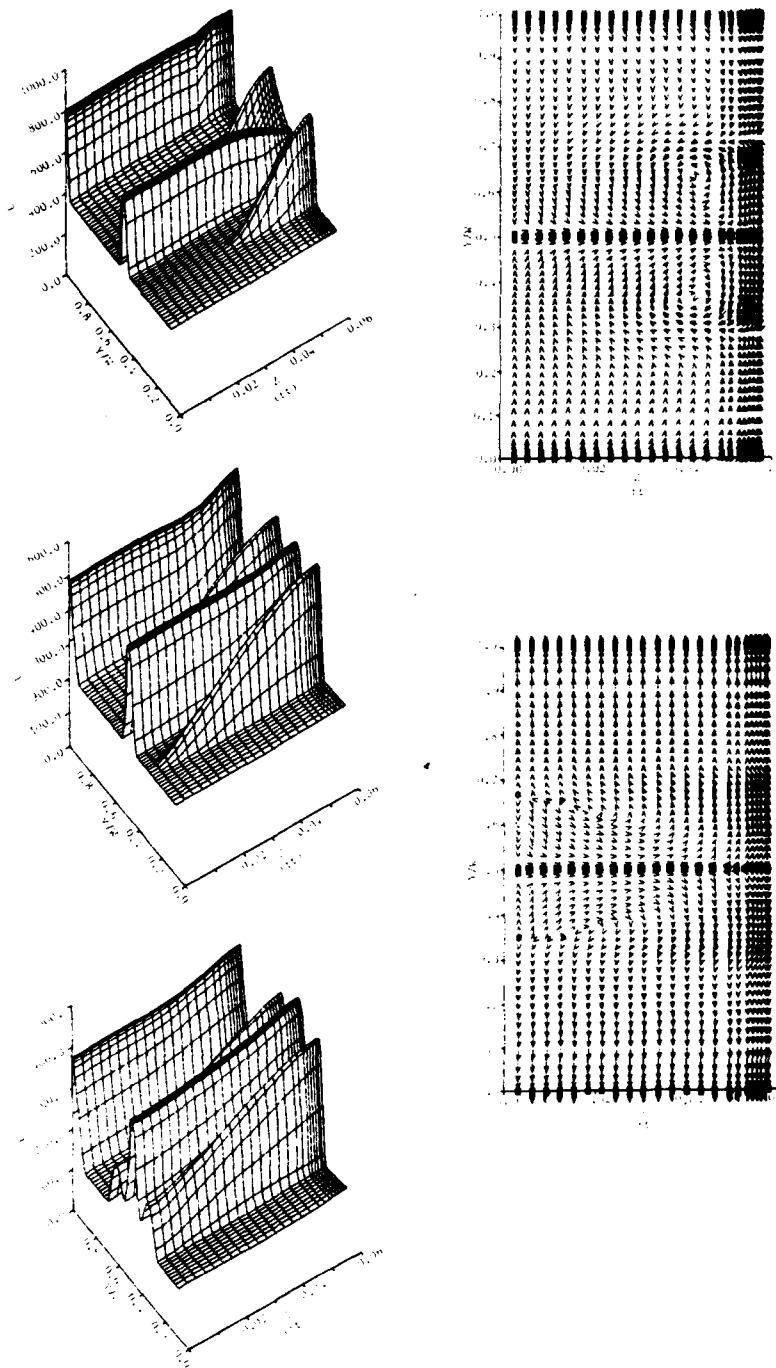
Figure 7-38   Bowtie Ratio 2 Nozzle

Rockwell International
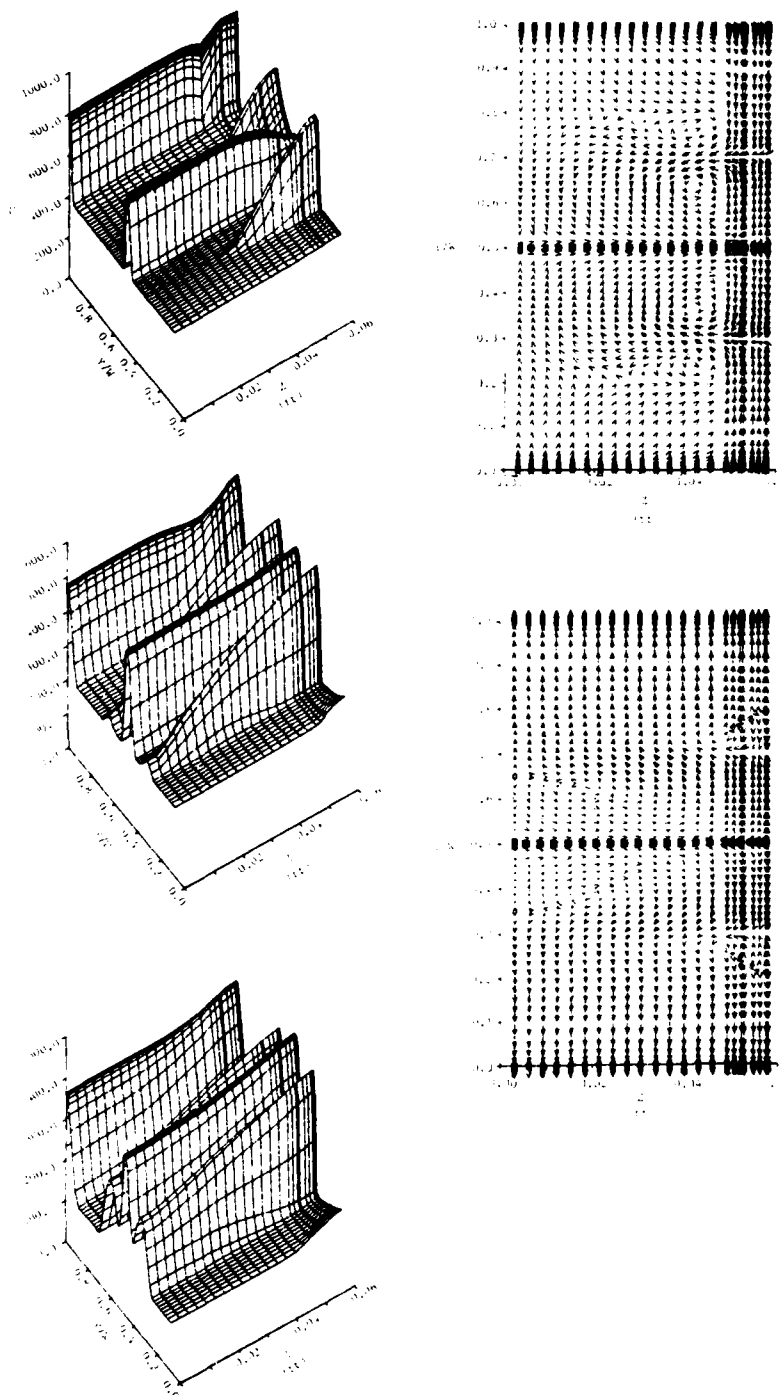
Figure 7-39   Bowtie Ratio 3 Nozzle

Rockwell International
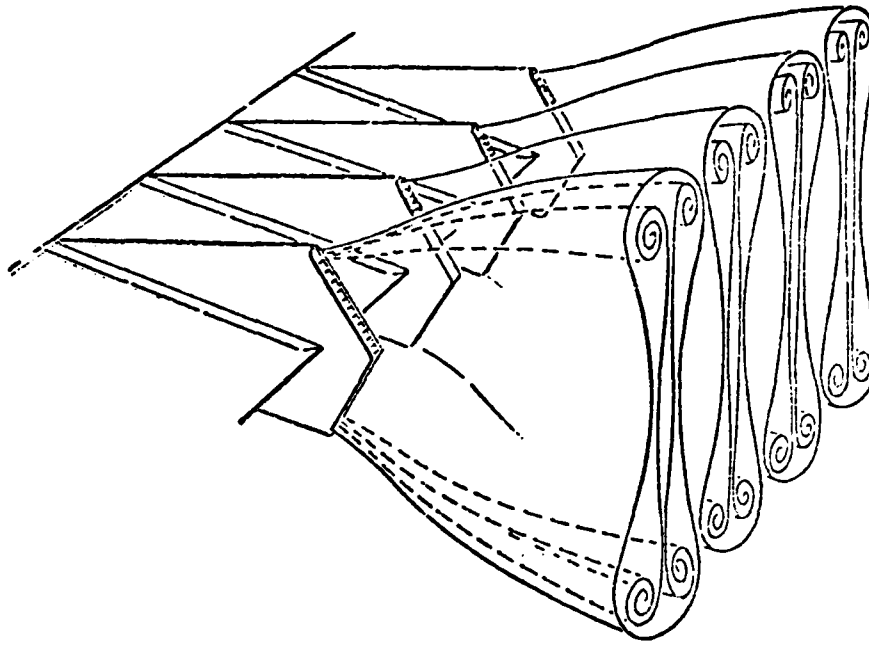
Figure 7-40   Simulation of Vortex Structure
Produced By Cross Slot Nozzle

7.4.3   Span Slot Aspect Ratio Effects

The increase in aspect ratio (length/thickness) of a slot nozzle in free
air is known to produce increased entrainment. Therefore, an experimental
and analytical study was conducted to determine the effect of span slot
aspect ratio on ejector performance.

The effect of span slot aspect ratio variations on ejector performance is
shown in Figure 7-43. The results show that reducing span slot aspect
ratio actually increased the augmentation ratio, which is contrary to
expectations based on free-air data.

Integration of the computed exit velocity profiles shown in Figures 7-44,
7-45, 7-46, and 7-47 reveals a three percent increase in massflow ratio
when span slot aspect ratio is reduced from 17.8 to 12. It is concluded,
that the larger aspect ratio span slots interfere with the development of
the cross-slot jets and thereby reduce the entrainment of the cross-slots.

Rockwell International
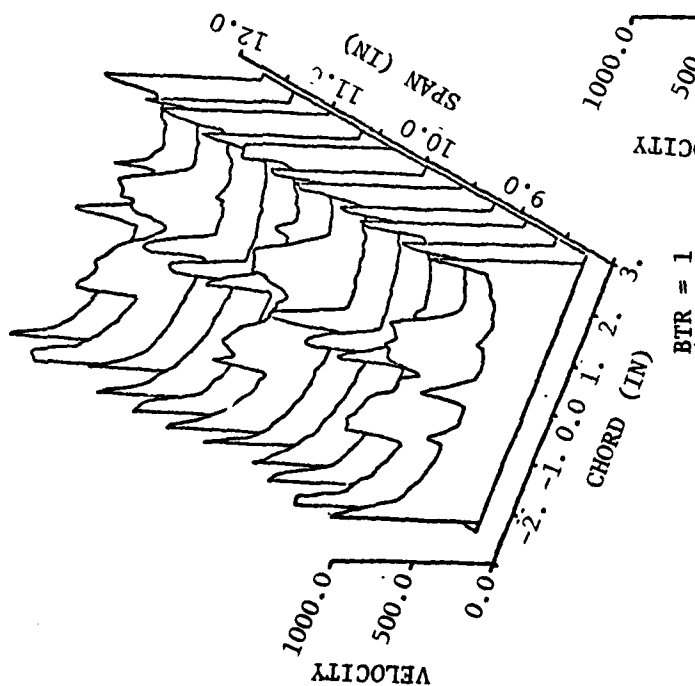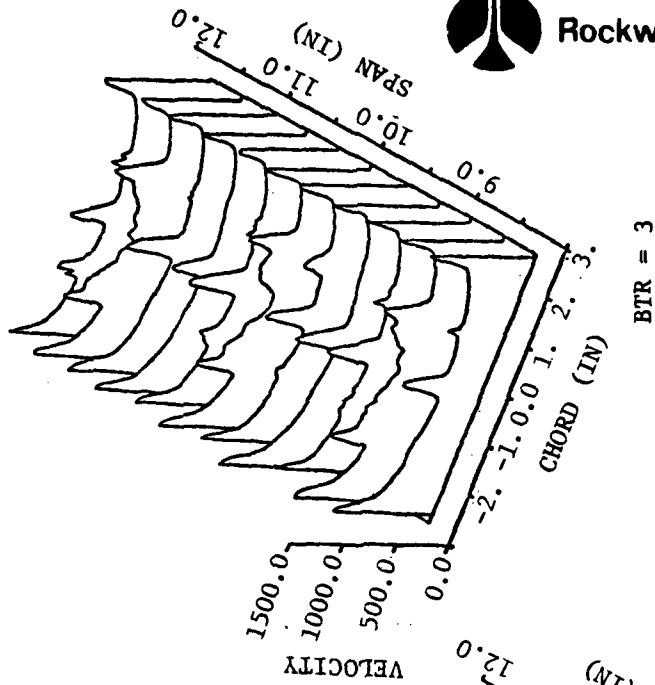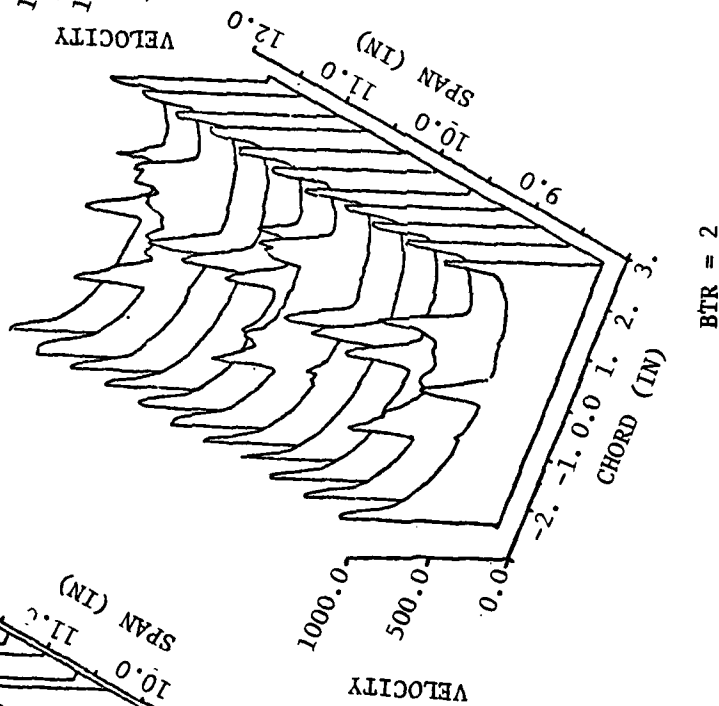
Figure 7-41 . Axial Velocity Profiles at Augmenter Throat

Rockwell International
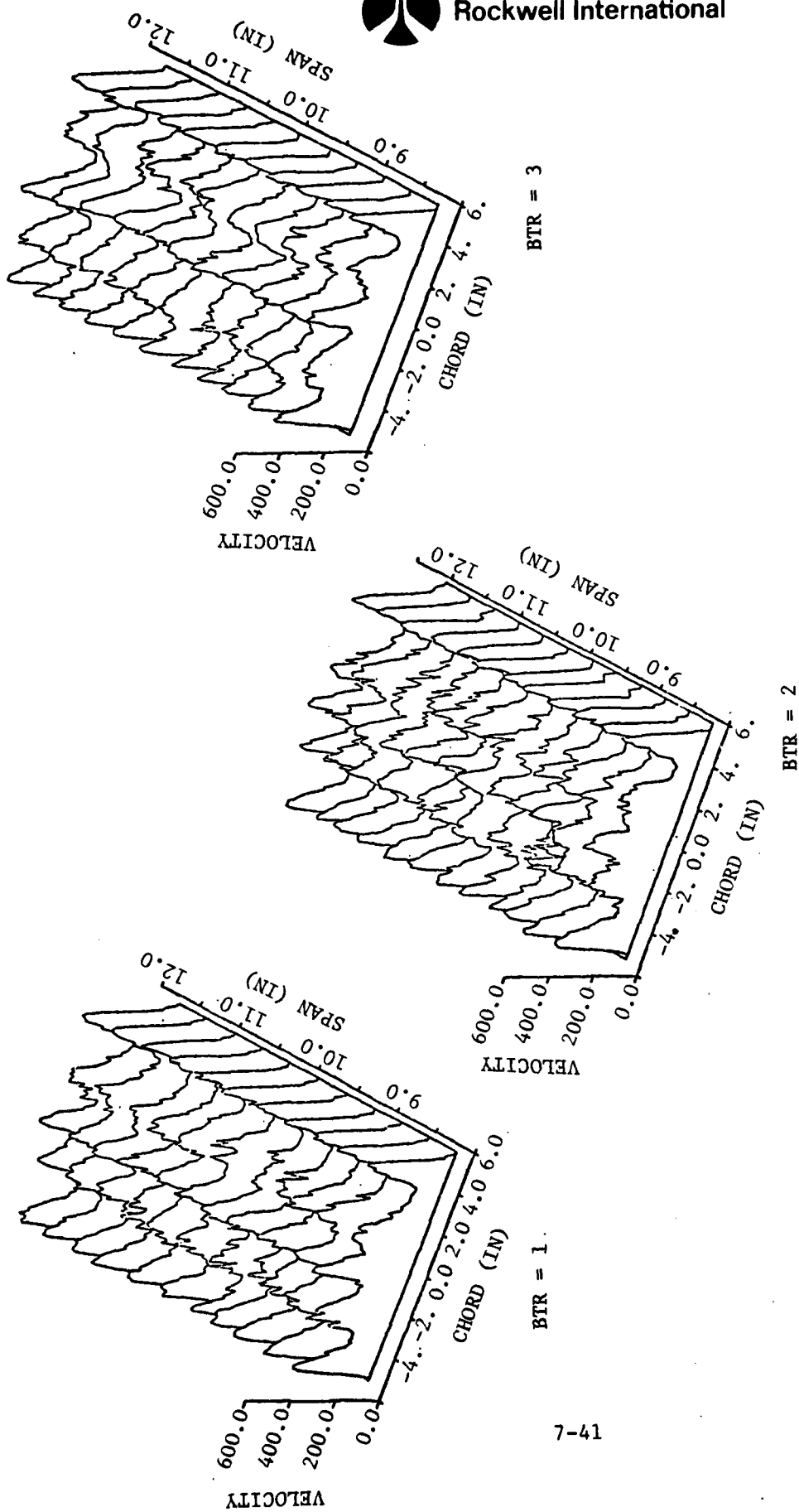
Figure 7-42. Axial Velocity Profiles at Augmenter Exit

Rockwell International
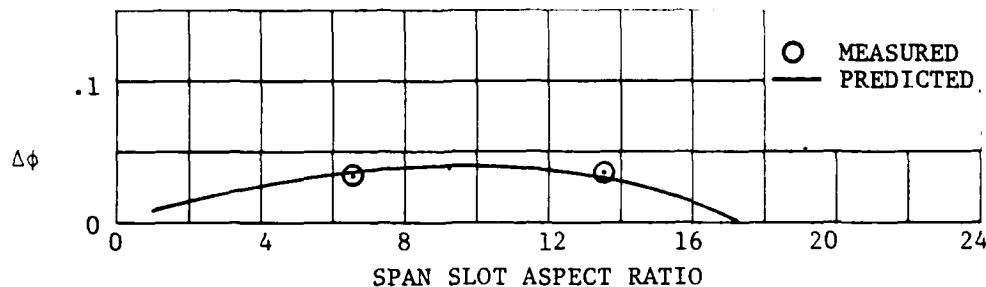
Figure 7-43 Effect of Span Slot Aspect Ratio On Thrust Augmentation

Therefore, the reduction of span slot aspect ratio results in an increase of entrainment by the cross-slots, that is greater than the loss due to reduced span slot aspect ratio.

7.4.4 Cross-Slot Aspect Ratio

The cross-slot jets play the dominant role in inducing flow through the ejector. Increased entrainment can be obtained by increasing the aspect ratio of the cross-slot jet. Therefore, an analytical study was conducted to determine the effect of cross-slot jet aspect ratio on ejector performance.

The jet mixing analysis code was used during the study. The baseline nozzle had a BTR=1 and 36 percent of the flow in the span slot. The diffuser area ratio was 2, and the initial jet angle was invariant with cross-slot aspect ratio. The cross-slot area remained constant as aspect ratio was varied. Details of this study can be found in Reference (27).

Figure 7-48 presents the predicted variation of incremental augmentation ratio with cross-slot aspect ratio. As expected, the results show a significant increase in augmentation as the aspect ratio is increased from 25 to 50. The variation of local exit massflow and total massflow is shown in Figure 7-49. The exit velocity profile shape factor, β , is shown in Figure 7-50. It can be seen that the increased aspect ratio has increased entrainment and produced flatter exit velocity profiles.

7.4.5 Effect of Span Slot/Cross-Slot Flow Split

An important nozzle design parameter is the division of total centerbody primary flow between the cross-slots and the span slots. Therefore, an analytical study was conducted to determine the optimum flow split for a fixed centerbody primary area and a fixed shroud geometry. Results of the study were verified by experimental data.
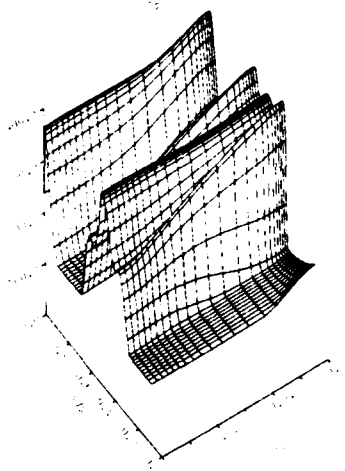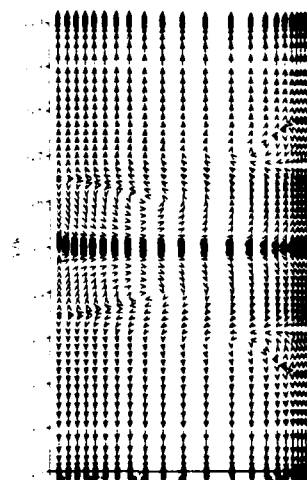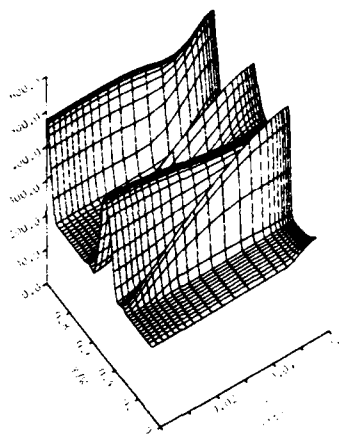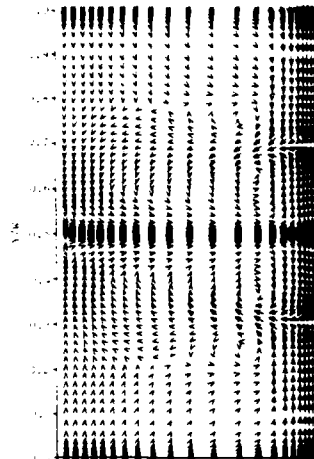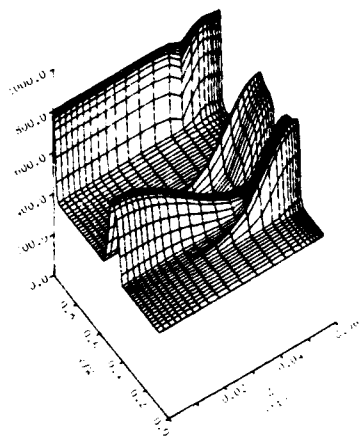
Figure 7-44   Span Slot Aspect Ratio 1
Nozzle

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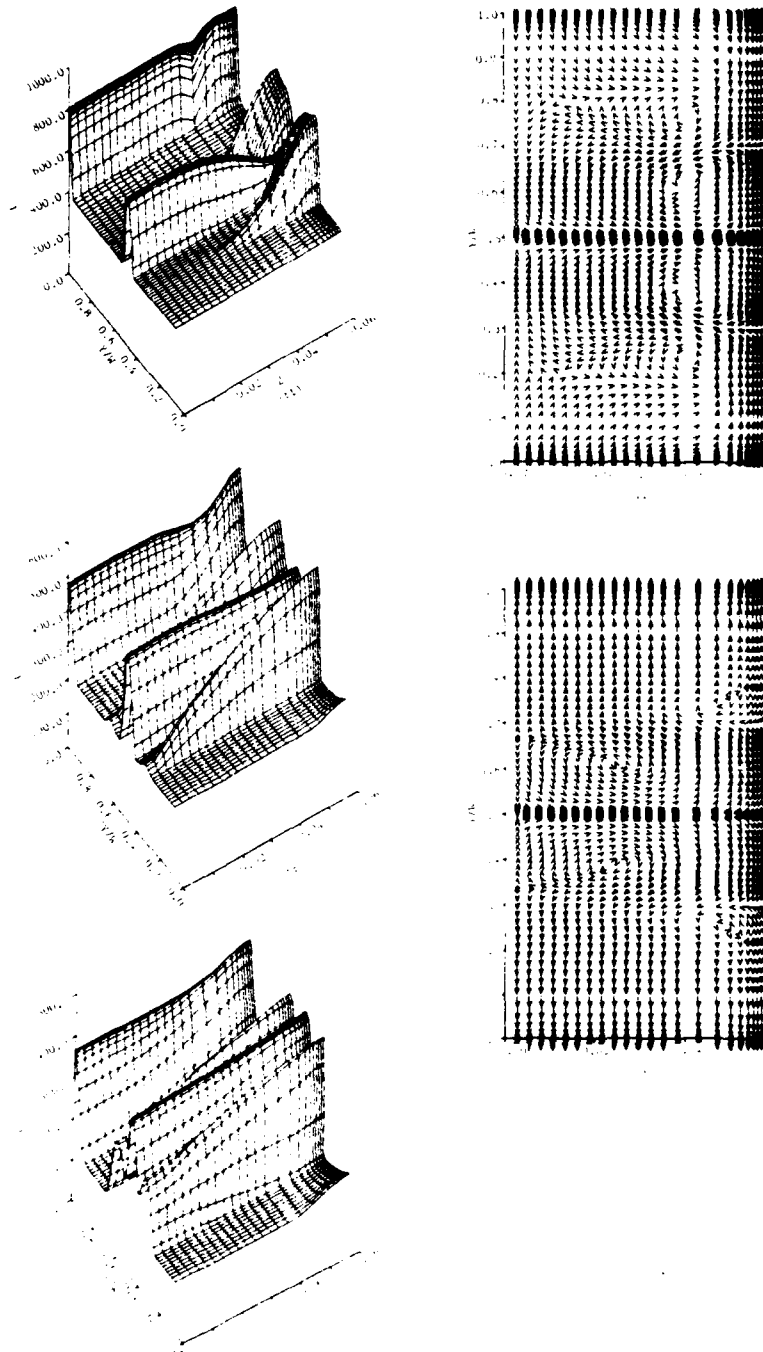
Figure 7-45   Span Slot Aspect Ratio 9
Nozzle

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Figure 7-46 Span Slot Aspect Ratio 12
Nozzle
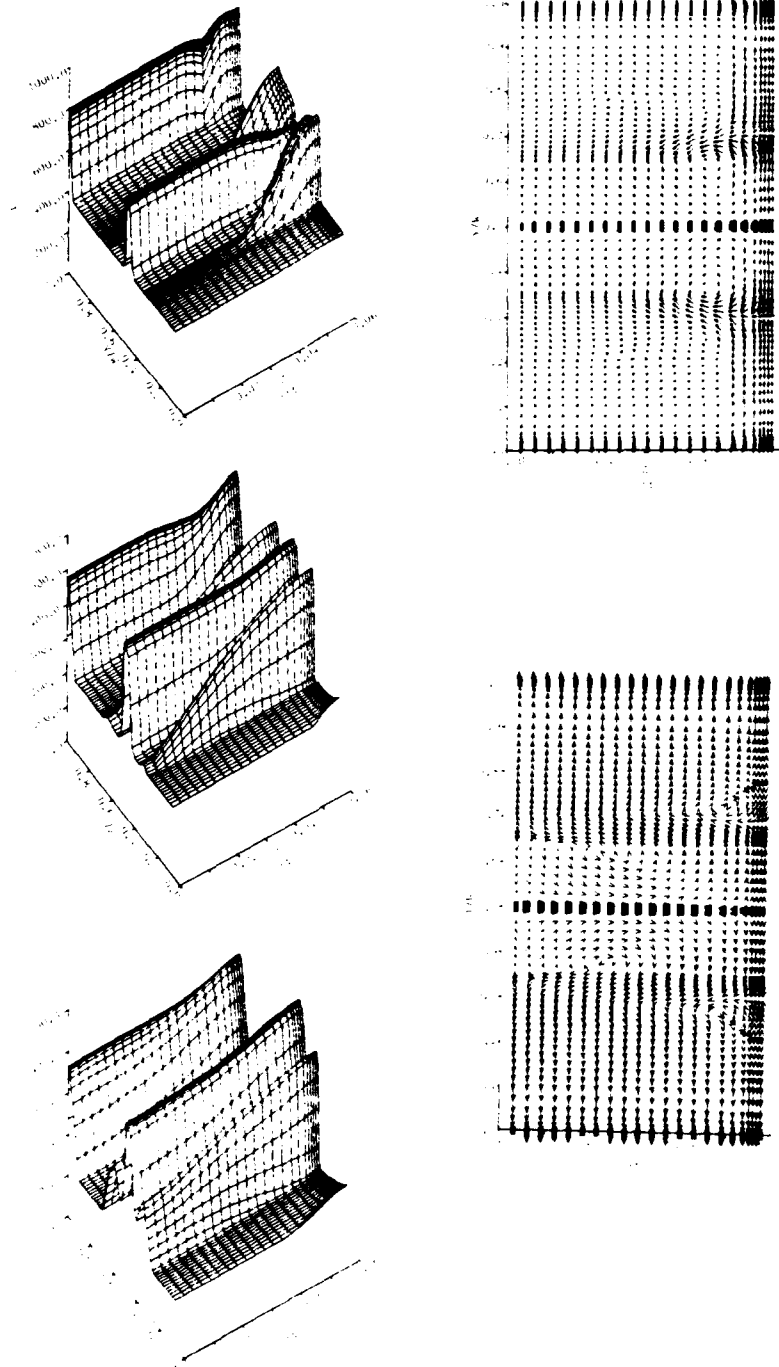
Figure 7-47   Span Slot Aspect Ratio 17.5
               Nozzle

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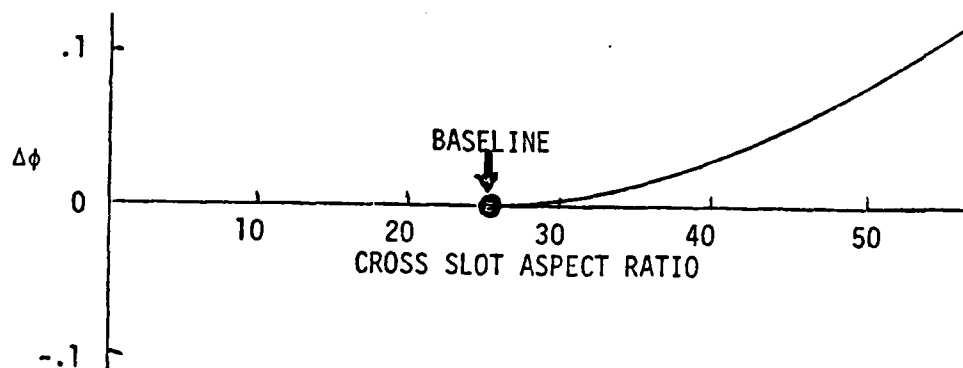
Figure 7-48 Effect of Cross Slot Aspect Ratio on Thrust Augmentation

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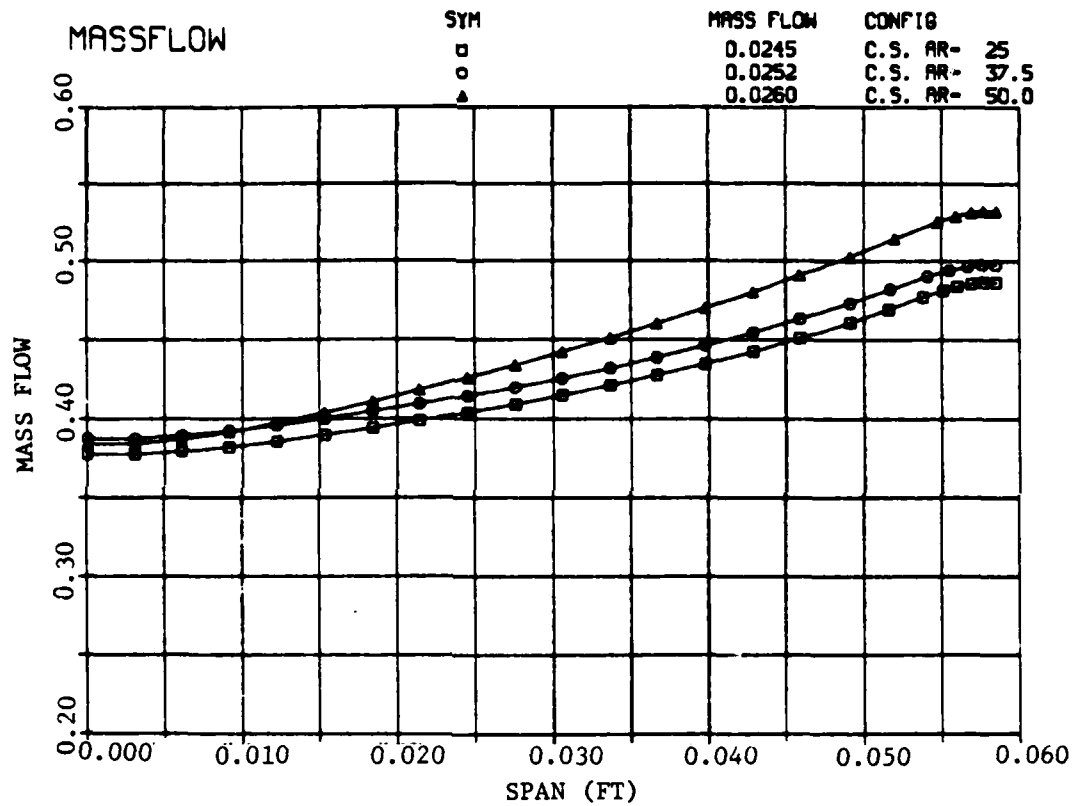
Figure 7-49 Spanwise Variation of Exit Massflow

Rockwell International
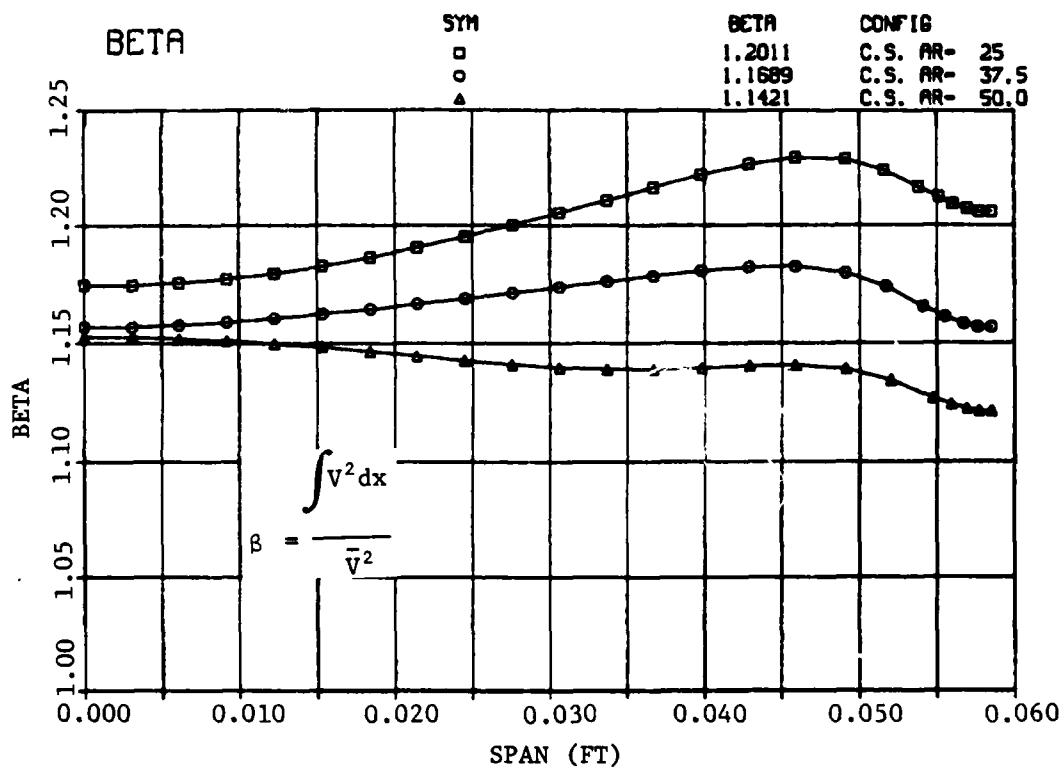
Figure 7-50   Spanwise Variation of Exit Velocity Profile Shape Factor, β

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The flow split is defined as the ratio of the total span slot flow area to the total centerbody flow area. A 100 percent flow split would indicate an all span slot configuration, while zero percent flow split would be all cross-slot. The flow split was obtained by varying the cross-slot or span slot thickness while maintaining a constant nozzle length.

The variation of incremental augmentation ratio with flow split is shown in Figure 7-51. As expected, the plane slot nozzle is the worst case; however, unexpectedly the augmentation increases as flow split is changed to an all cross-slot configuration.

The jet development for flow splits of 0%, 60%, and 80% are shown in Figures 7-52, 7-53, and 7-54, respectively. The 36 percent flow split was shown previously in Figure 7-37. Referring to the exit axial velocity plots, it can be seen that the span slots apparently impede the development of the cross-slot jets and hence limit the entrainment potential.

The variation of total massflow at the ejector exit is shown in Figure 7-55 for flow splits of 0%, 36%, and 60%. The corresponding velocity profile shape factors, β, are shown in Figure 7-56.

Verification of the flow split trend was obtained experimentally by designing and testing a nozzle with 65 percent of the flow to the span slot. The incremental augmentation is shown in Figure 7-51 and compares favorably with the predictions. However, it should be noted that the experimental results were obtained for BTR = 2 nozzles, whereas, the analysis was conducted with BTR = 1.0 nozzles. This difference is not believed to have an effect on the trends.

A comparison of the exit axial velocity profiles for the 65 percent and 36 percent flow split nozzles are shown in Figure 7-57. These experimental plots show clearly the influence the span slot has on the development of the cross-slot jet.

Further experimental verification of the effect of flow split was obtained by designing and testing an all cross-slot nozzle. The nozzle was designed to have a bowtie ratio = 1. The all cross-slot nozzle results shown in Figure 7-51 indicate that it is slightly better than the baseline nozzle (BTR = 1). A comparison of the exit axial velocity profiles for zero percent and 36 percent flow split nozzles are shown in Figure 7-58. This result is in qualitative agreement with the analysis.

It is concluded from both experiment and analysis that flow splits between zero percent and 50 percent have little effect on thrust augmentation.

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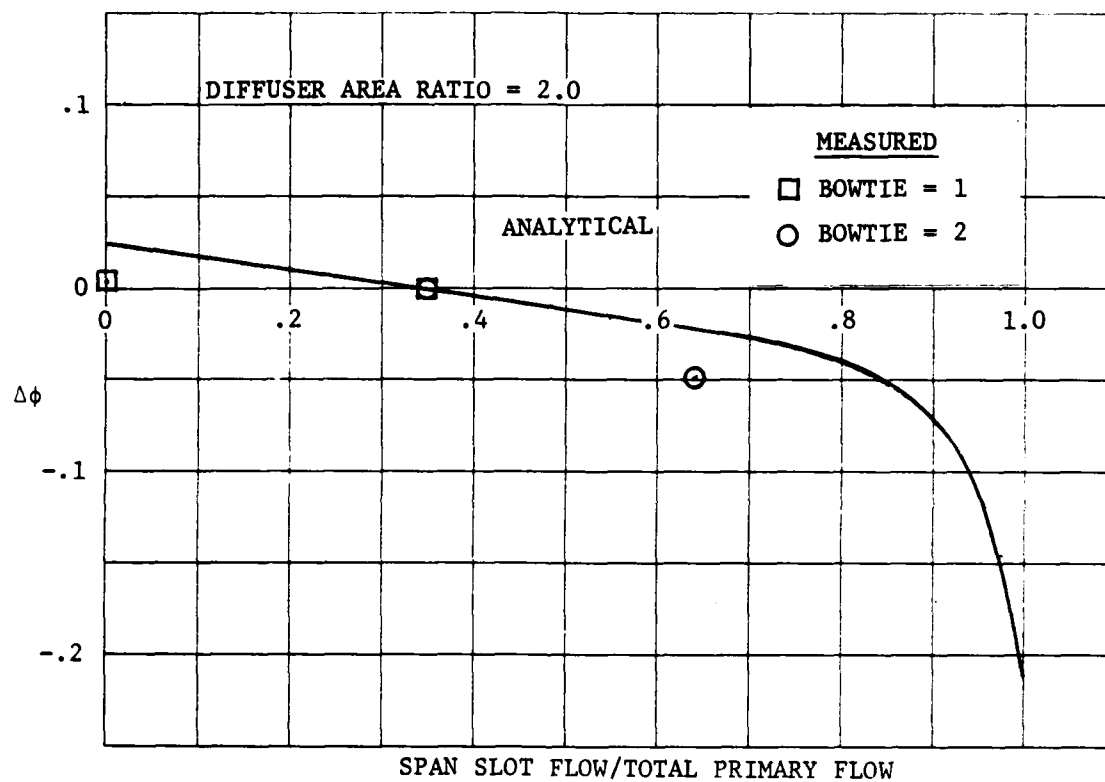
Figure 7-51 Effect of Flow Split On Thrust Augmentation
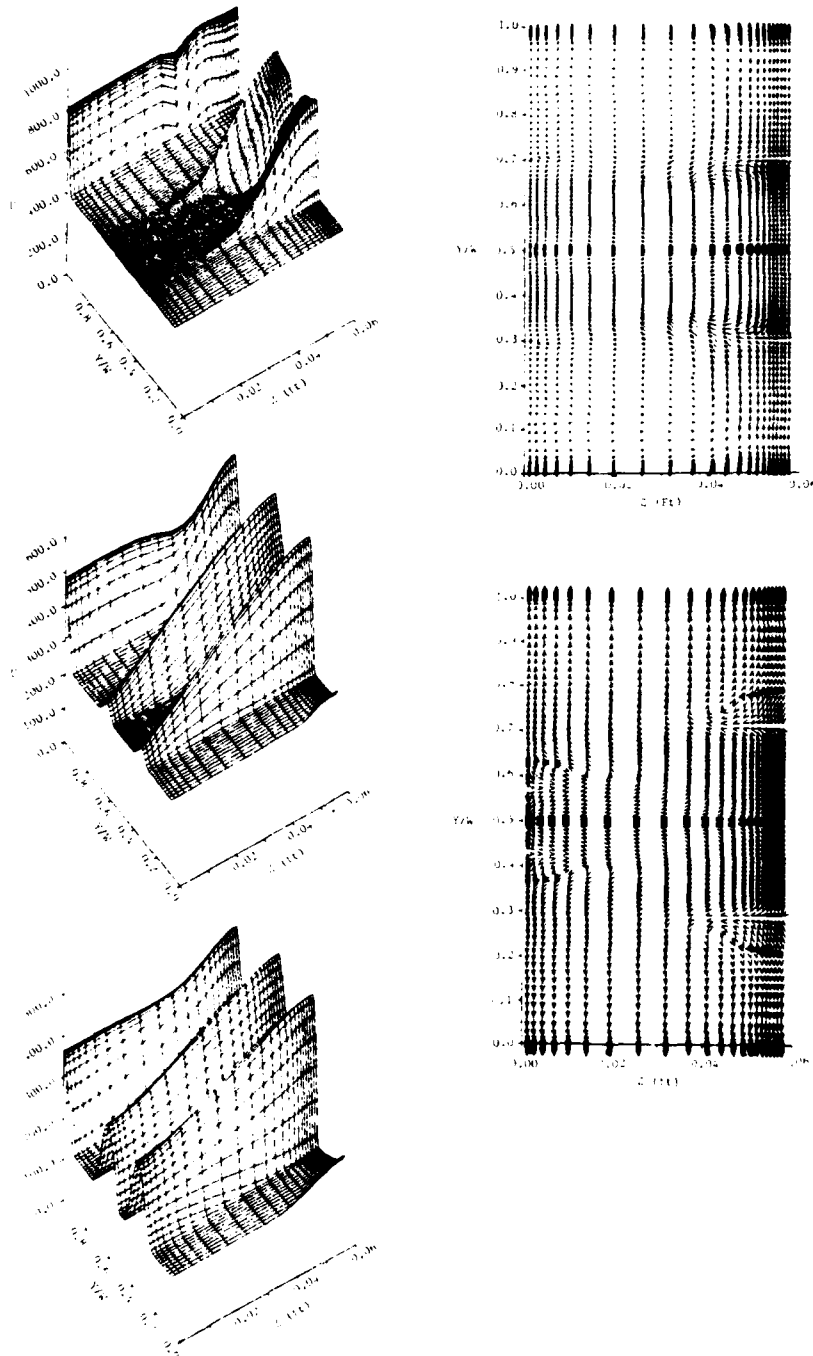
Figure 7-52   0% Flow Split Nozzle

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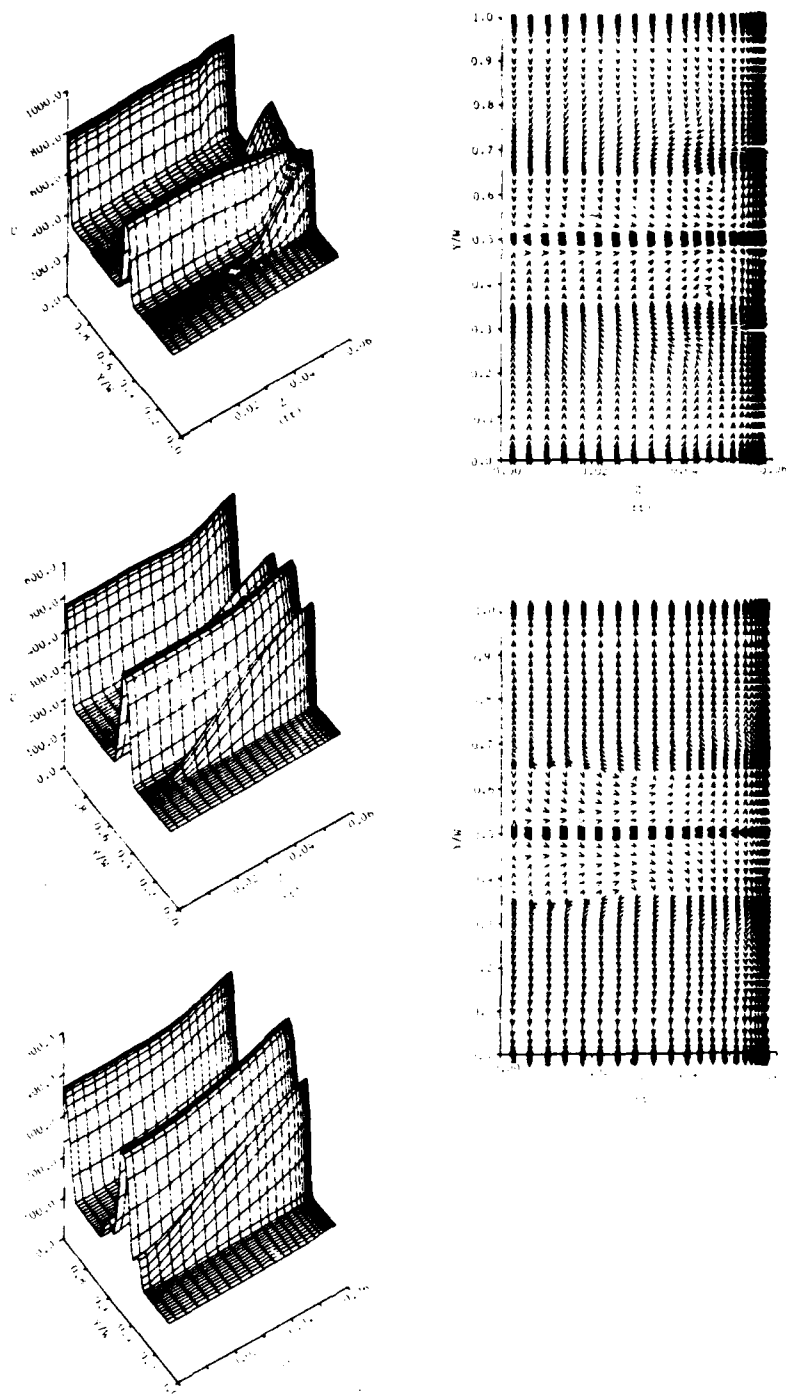
Figure 7-53   60% Flow Split Nozzle

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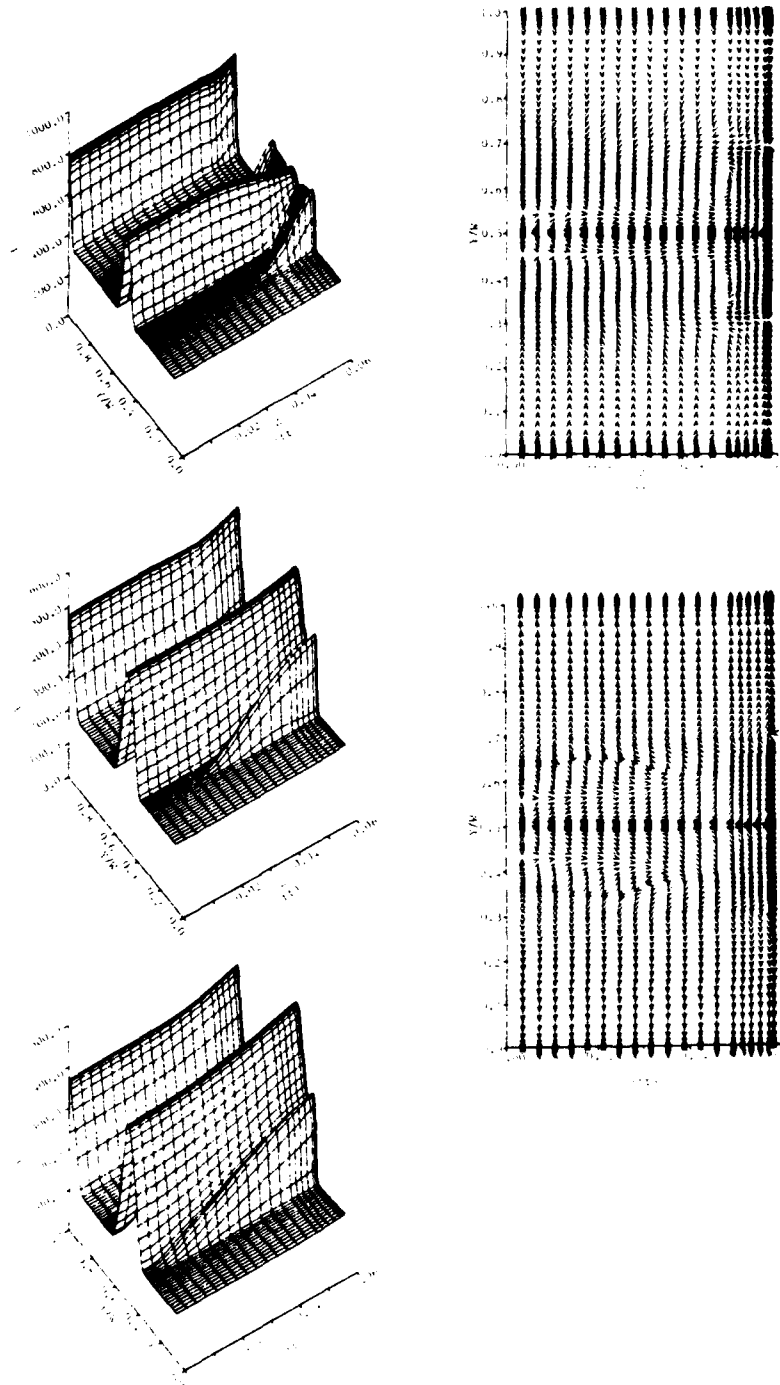
Figure 7-54   80% Flow Split Nozzle

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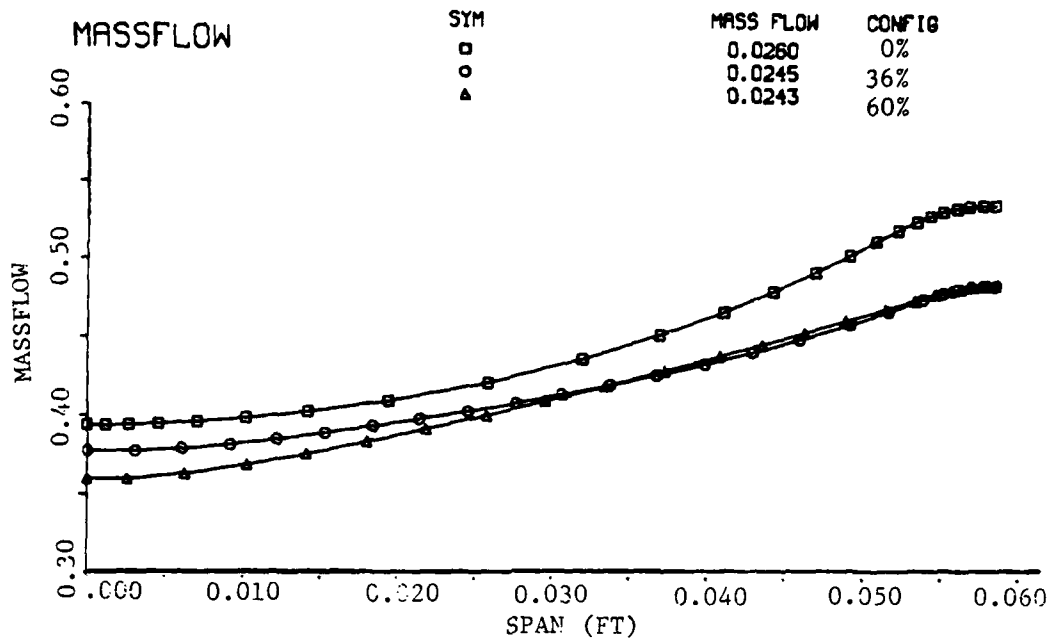
Figure 7-55 Effect of Flow Split on Spanwise Variation
of Exit Massflow
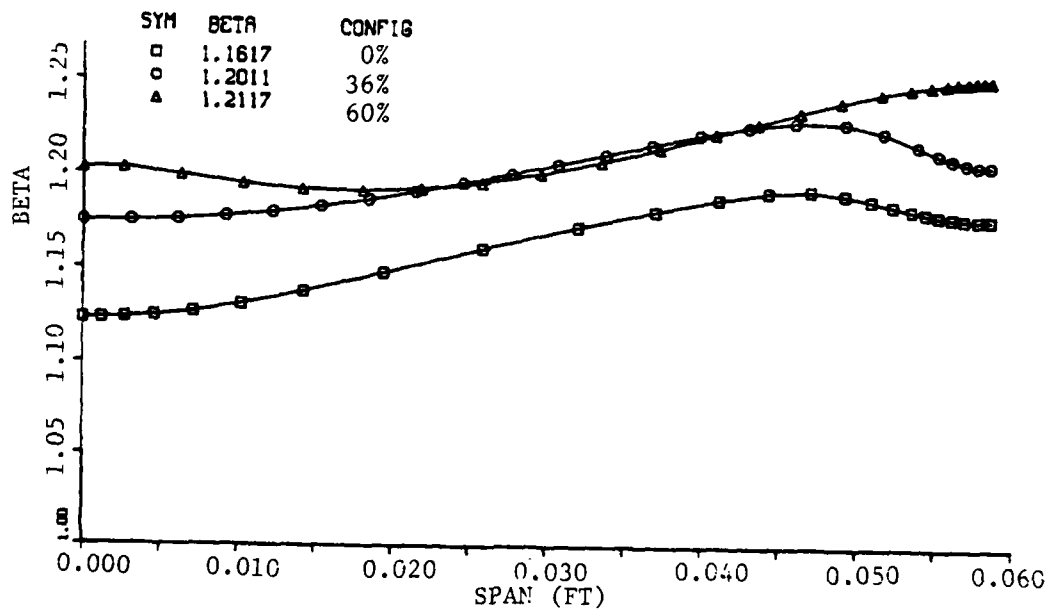
Figure 7-56 Effect of Flow Split On Spanwise Variation of
Exit Velocity Profile Shape Factor

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65% FLOW SPLIT
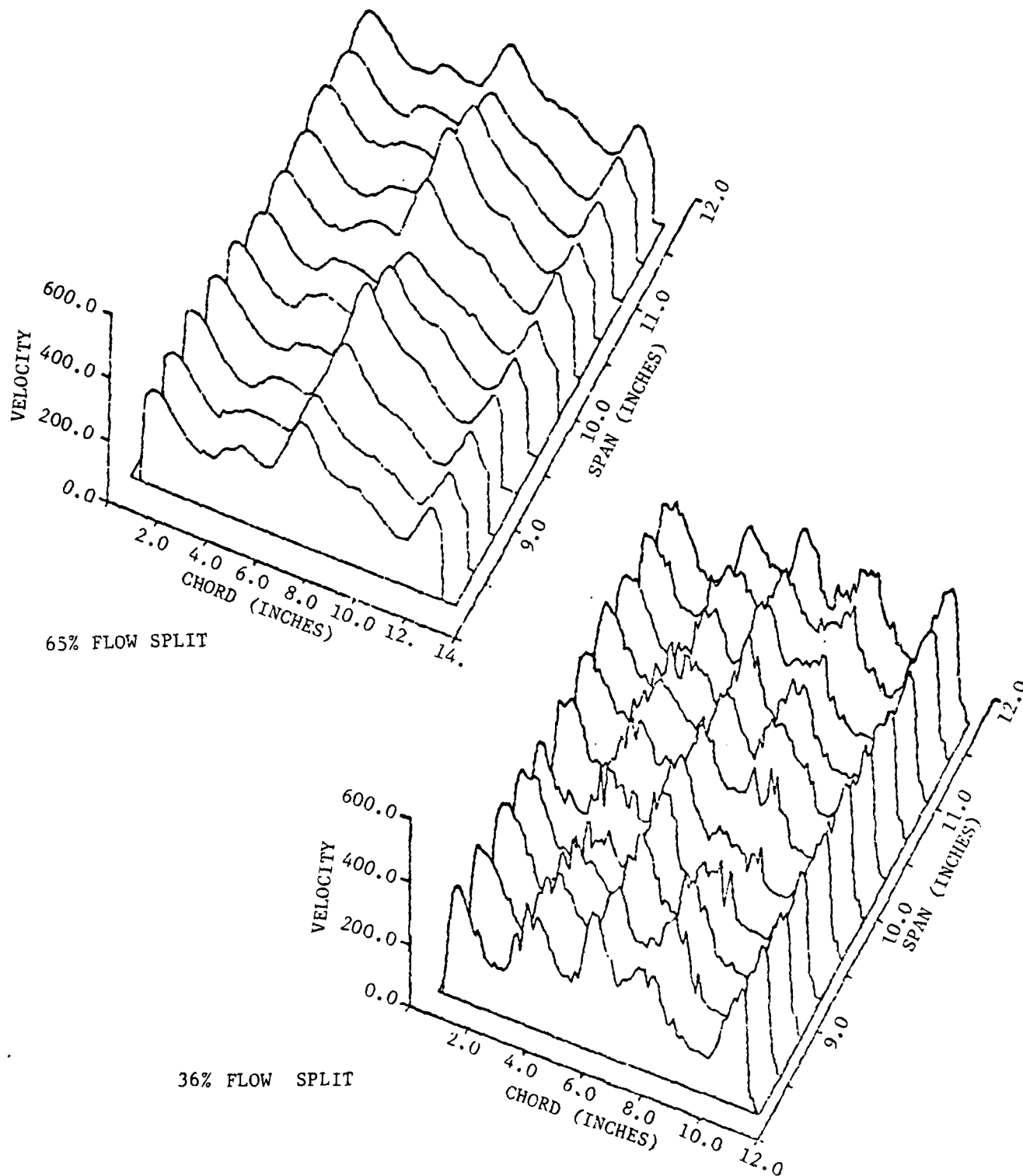
36% FLOW SPLIT

Figure 7-57 Comparison of Exit Axial Velocity Profiles for 36 Percent
and 65 Percent Flow Split Nozzles

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0% FLOW SPLIT
(ALL CROSS-SLOT)
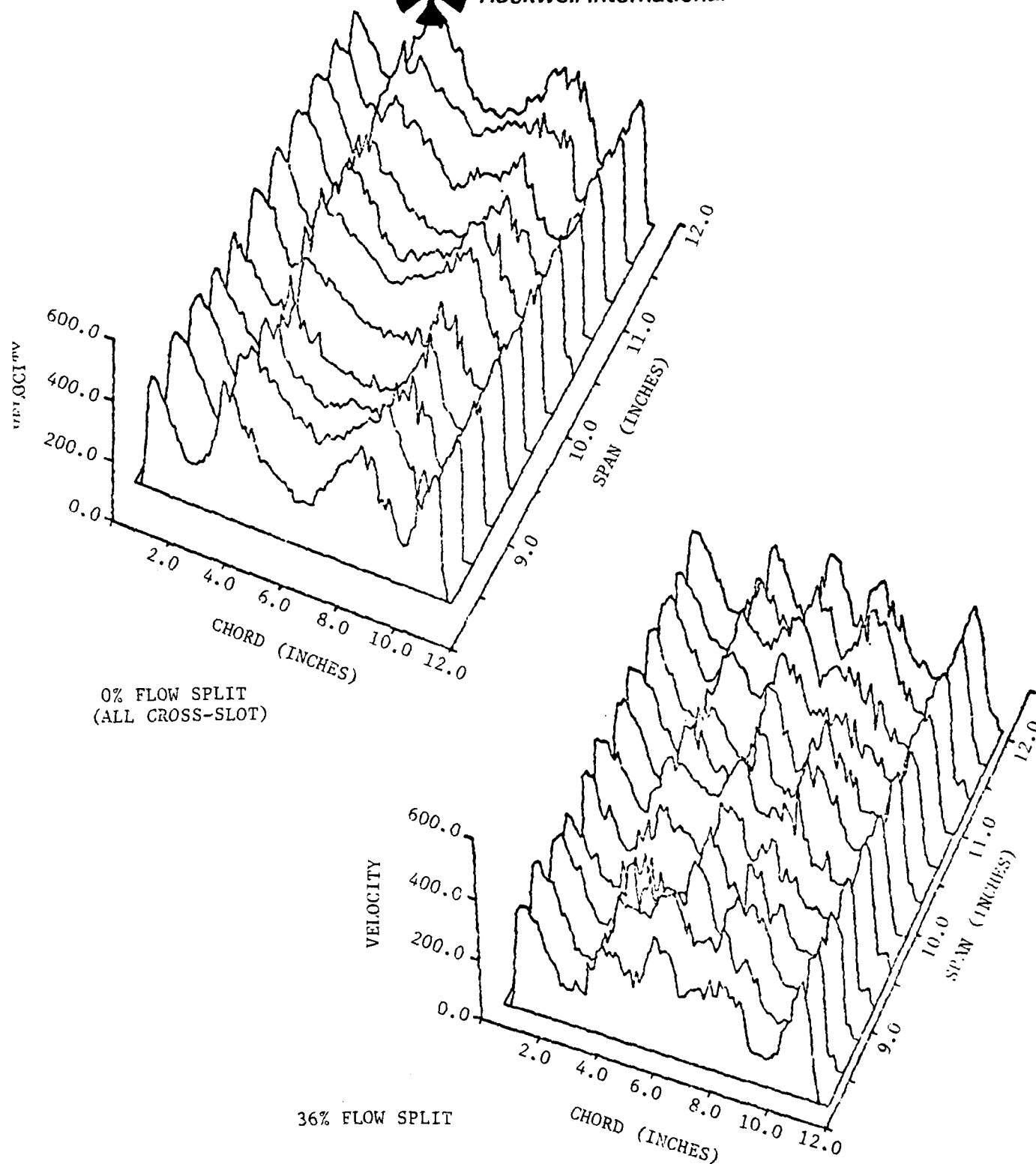
36% FLOW SPLIT

Figure 7-58 Comparison of Exit Axial Velocity Profiles for 0 Percent
and 36 Percent Flow Split Nozzles

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7.4.6 Effect of Number of Elements

The effect of increasing the number of nozzle elements within a fixed span was evaluated analytically. The study was conducted at diffuser area ratio = 2, BTR = 1, 36 percent flow split and fixed total nozzle area. The configurations evaluated were obtained by fixing the cross-slot width and the span slot thickness and varying the cross-slot thickness and span slot length proportionately.

Figure 7-59 shows that increasing the number of elements results in significant increases in augmentation. This is a consequence of the increased cross-slot aspect ratio and reduced span slot aspect ratio as shown in Figure 7-60. These results are consistent with those previously presented for aspect ratio.
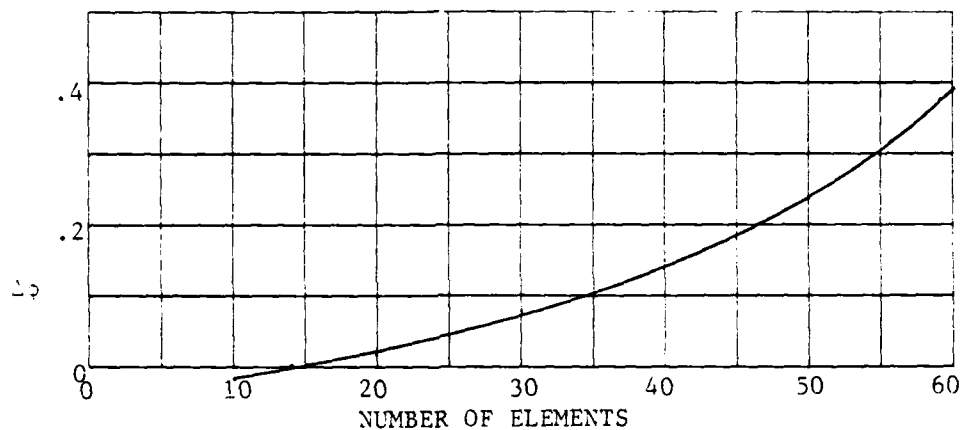
Figure 7-59   Variation of Thrust Augmentation With Number of
Elements (C_T = Constant)

The analytical results were obtained using the same nozzle thrust coefficient for all configurations. However, experimental results for asymmetric nozzles presented in Reference (28) show a reduction in thrust coefficient as the number of elements is increased. Adjusting the analytical values shown in Figure 7-59 for C_T effects produces the results of Figure 7-61. Therefore, it is concluded that increased mixing obtained by increasing the number of elements would be partically compensated by a loss in thrust coefficient.

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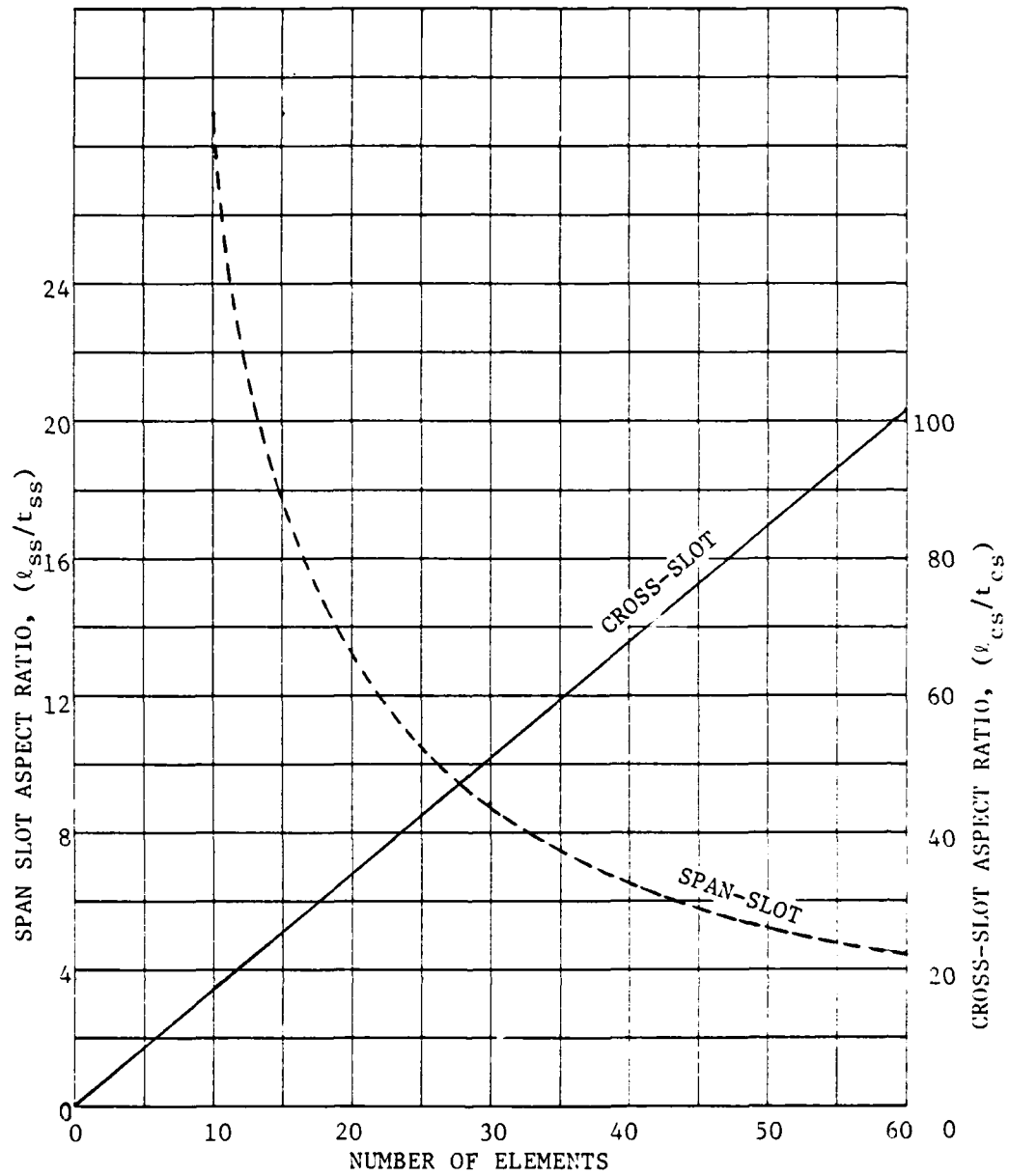
Figure 7-60 Variation of Nozzle Aspect Ratio With Element Spacing

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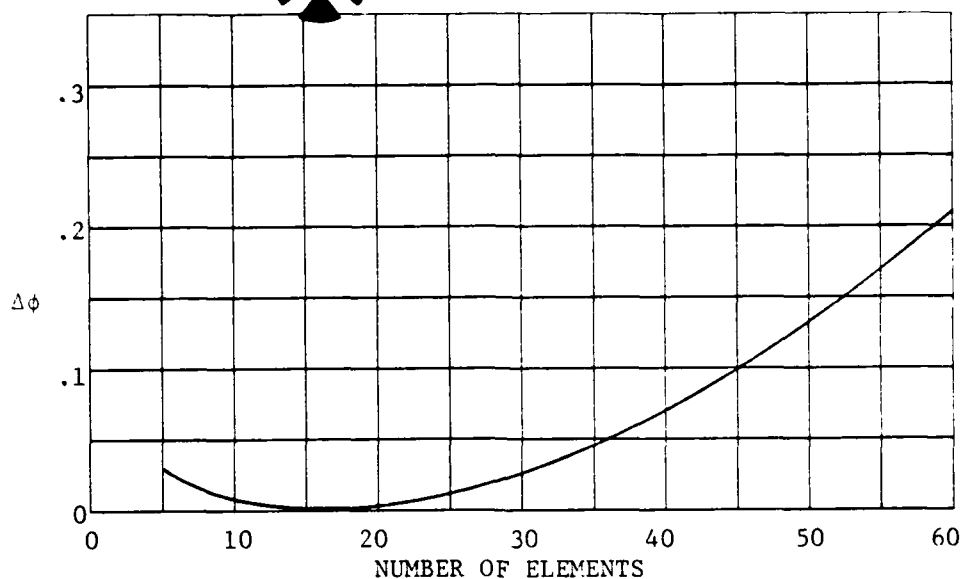
Figure 7-61 Variation of Thrust Augmentation With Number of Elements
(Adjusted for C_T Effects)

7.4.7 Jet Angle Study

The hypermixing nozzle studies, both test and analysis, had identified
that the initial jet flow angle was a primary factor in ejector
performance. The flow angles generated by the hypermixing nozzles were
directly related to the angle of the slots and were easily known, a
priori, for both tests and analysis. Correlation and/or nozzle design was
therefore rather straightforward. However, the cross slot nozzles
presented a problem in that the relationship between flow angle and
geometry was not usually known. Therefore, an experimental test matrix
was defined for the purpose of determining the dependence of flow angle on
nozzle geometry. A parallel analytical study was conducted (using the jet
mixing analysis) to determine the effect of jet flow angle on augmentation
ratio. Correlation of the two studies resulted in a capability to define
the nozzle geometry required to obtain the desired augmentation ratio.
The results of these studies are presented in this section.

Figure 7-62 presents the test matrix to determine the effects of cross
slot nozzle angles on the jet flow angles. This matrix was designed to
study the effects of launch angle at constant wedge and the effect of
wedge angle at constant launch angle. Four centerbodies were designed and
tested. Free air surveys were made using a five-port flow direction probe.

The results of the tests are shown in Figure 7-63 where measured flow
angles are shown plotted versus non-dimensional distance from the center
of the cross slot. Comparison of Figure 7-63a, 7-63b, and 7-63c shows that
the mean jet flow angle increases with wedge angle. Launch angle (Figure
7-63a, 7-63c, and 7-63d) has an effect on the angle distribution over the
last 20 percent of cross slot nozzle width.

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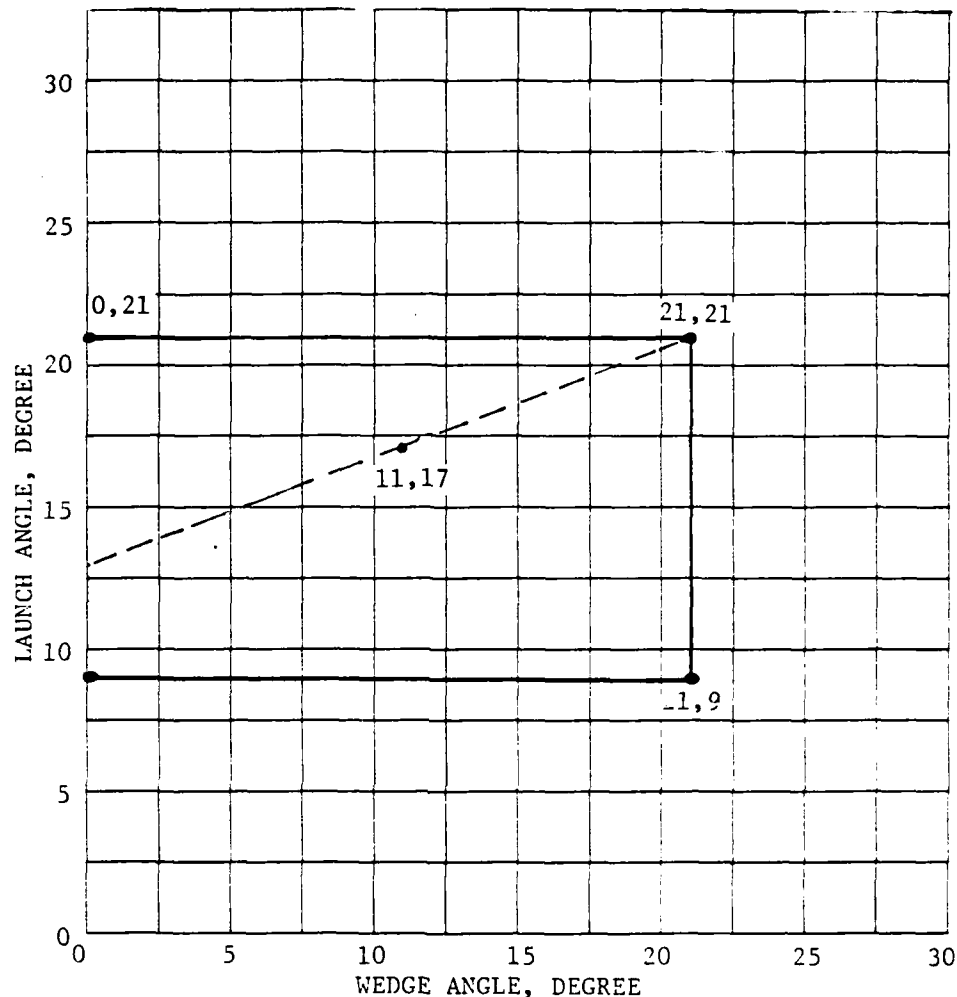
Figure 7-62   Nozzle Test Matrix for Jet Angle Study

The effect of cross-slot nozzle flow angle on ejector performance was obtained using the jet mixing analysis computer code. The analysis was conducted utilizing the .2 scale rectangular wing model geometry at an exit area ratio of 2.0. Analytical solutions were obtained for assumed angle distributions which varied linearly across the nozzle and for constant angle distributions. A comparison of a linear and a constant angle distribution is shown in Figure 7-64.

The computed augmentation ratios are shown in Figure 7-65 as a function of average flow angle. The results indicate that augmentation is a function of both angle distribution and magnitude. The peak augmentation for both types of distributions appears to occur at about $\alpha_{avg} = 18°$.

The computed development of the jets for angle distributions of (0,0), (0,28), and (21,21) are shown in Figures 7-66, 7-67, and 7-68, respectively.

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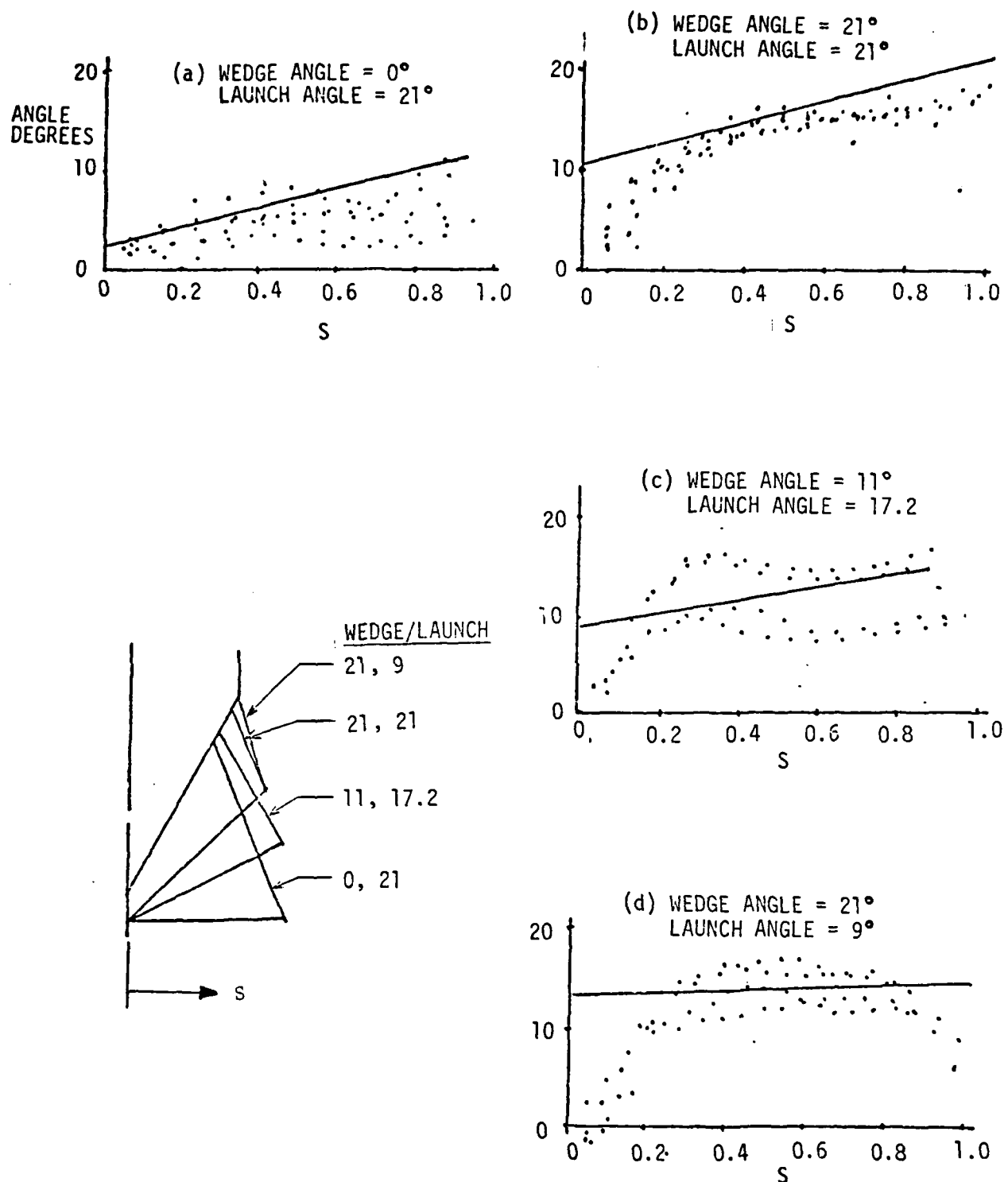
Figure 7-63 Measured Cross-Slot Nozzle Exit Flow Angles

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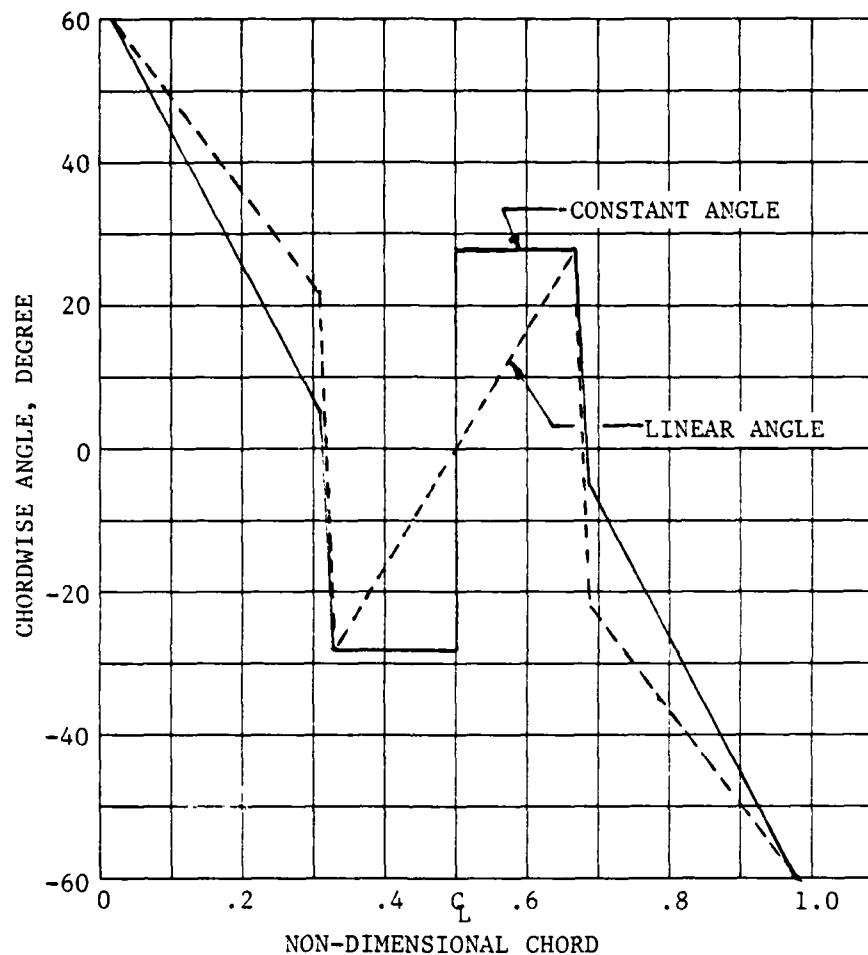
Figure 7-64 Assumed Angle Distribution at Initial Data Plane

Comparison of the axial velocity plots at the ejector exit show that
increased vortex strength due to jet angle has increased the mixing
(spreading) of the cross slot jets. Furthermore, the linear angle
distribution configuration promoted greater mixing as is apparent from the
elimination of the secondary velocity region between the span slot and the
cross slot.
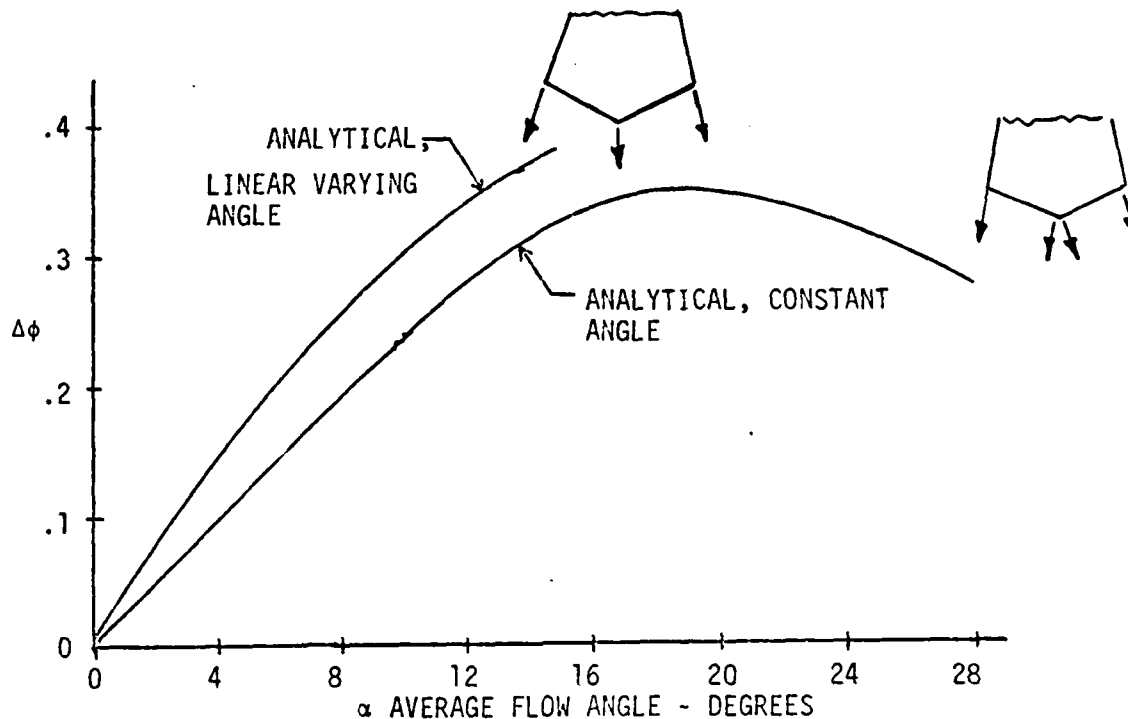
Figure 7-65   Effect of Constant and Linearly Varying Cross-Slot Jet Flow
Angles on Thrust Augmentation

The results of the combined experimental/analytical jet angle study were
utilized to design a nozzle capable of a linear jet angle variation. A
sketch of the nozzle is shown in Figure 7-69 and the nozzle is defined in
Reference (29). The nozzle had a BTR = 1, flow split = 36 percent, and
linear angle variation from zero degrees to 37.2 degrees at the tip.

Free-air measurements of the nozzle flow angularity is shown in Figure
7-70. A nearly linear angle variation was obtained and indicates the
capability to design a nozzle based on desired flow characteristics.

The performance of this nozzle is compared in Figure 7-71 with a nozzle
having a nearly constant jet angle. The analytical results showed that
the linear angle nozzle should have produced an augmentation level of
about .05 greater than obtained experimentally. The results obtained with
this nozzle and several others led to a more detailed study of the shroud
surface pressures. The pressures indicated the possibility of local
separation on the Coanda surfaces for nozzle configurations capable of
thrust augmenation levels above 1.63. Therefore, separation effects make
it difficult to determine relative nozzle performance. These effects are
discussed in more detail in the shroud section.

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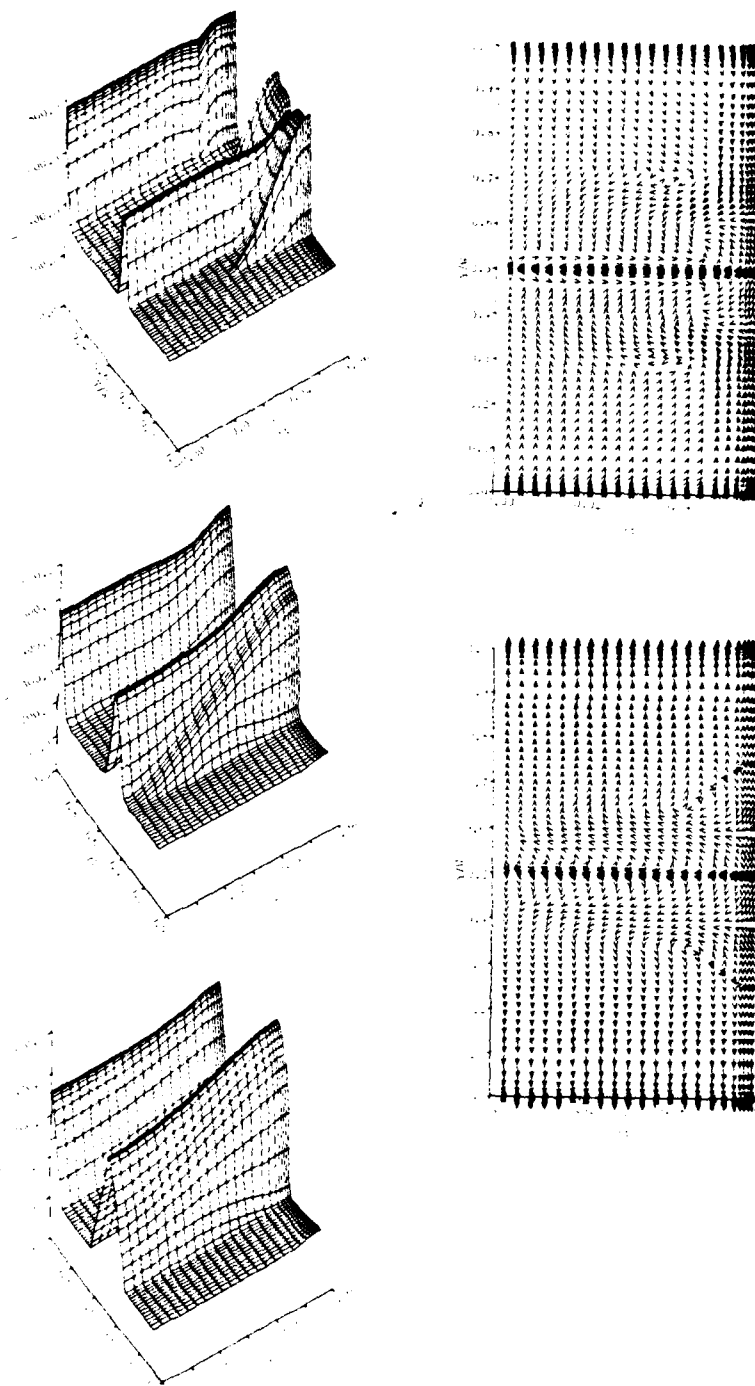
Figure 7-66   Zero Degree Jet Angle Nozzle

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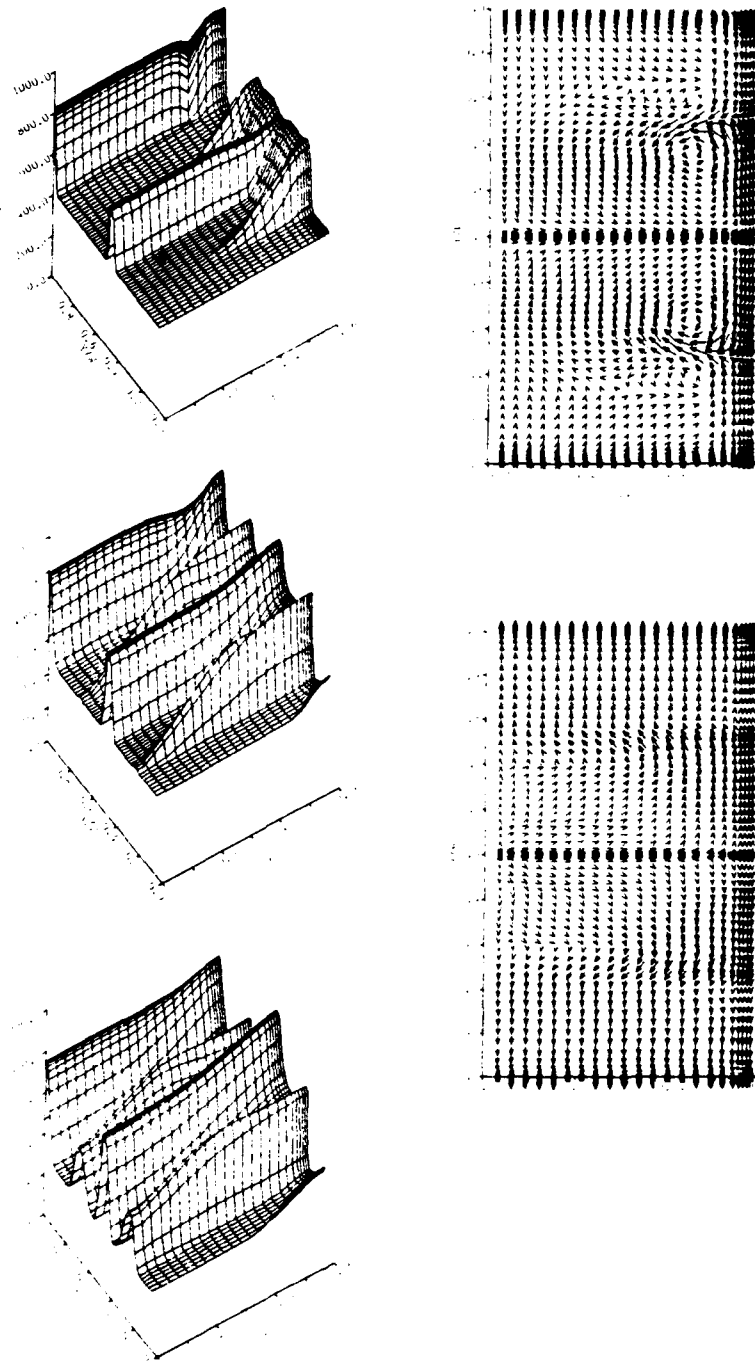
Figure 7-67   Linear Jet Angle
             (0 to 28 Degrees) Nozzle

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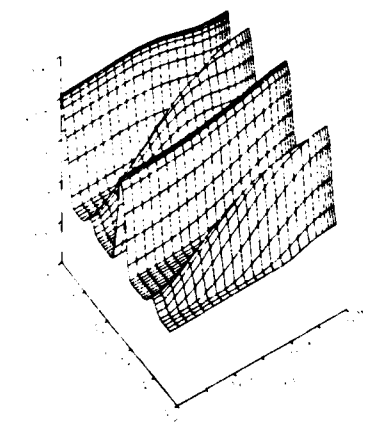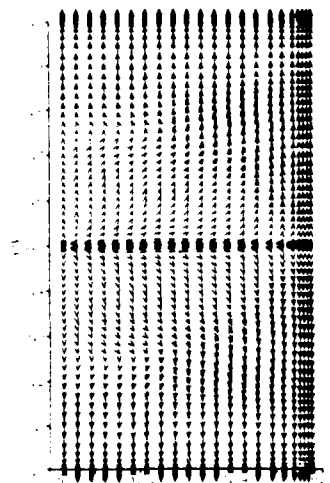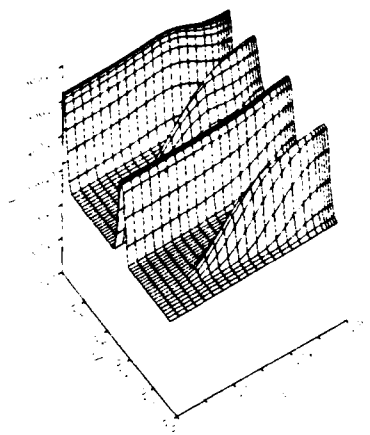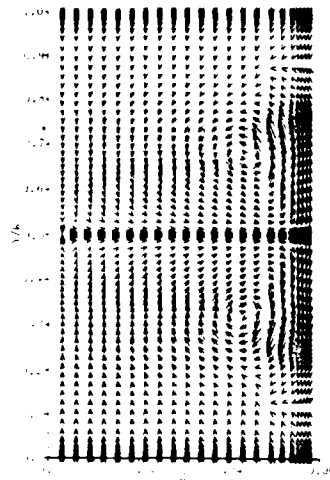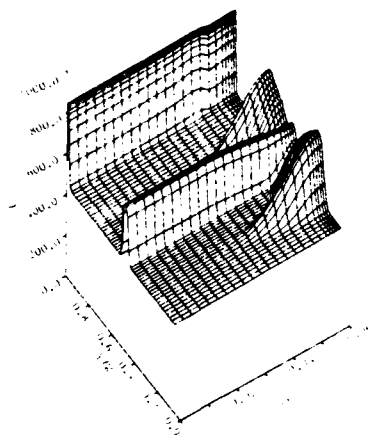
Figure 7-68   Constant Jet Angle
                (21 Degrees) Nozzle

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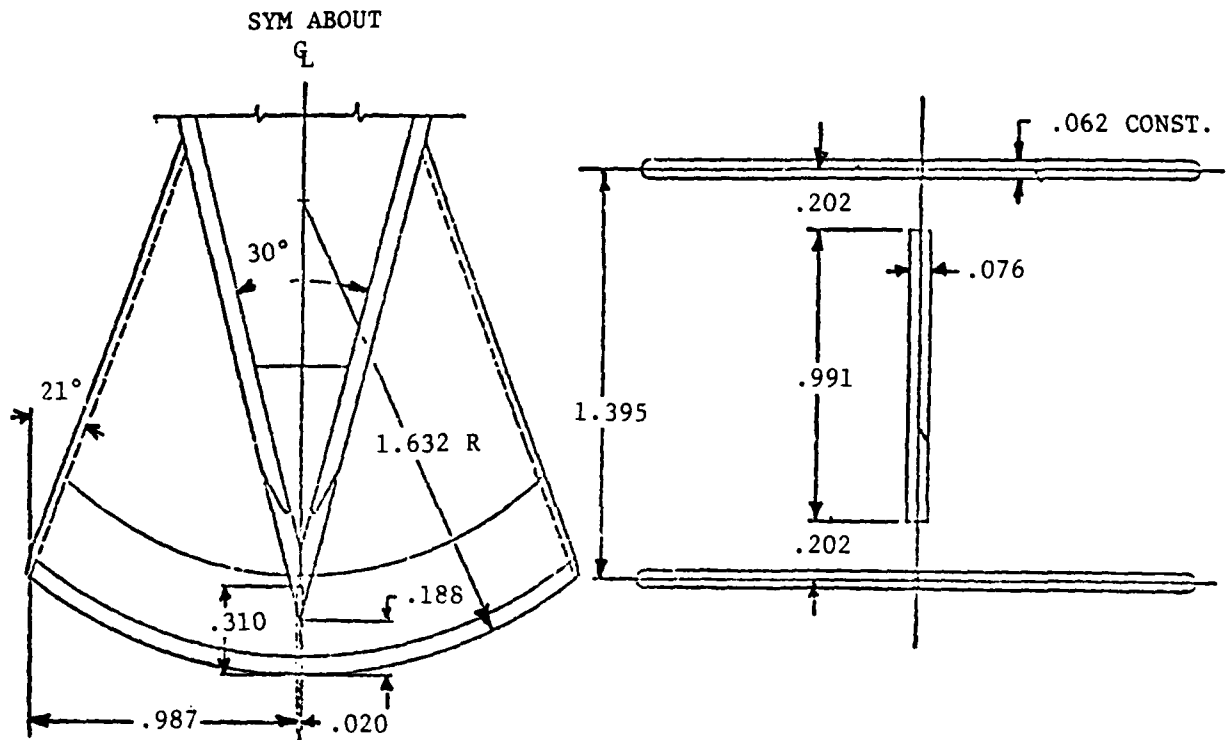
Figure 7-69   Cross Slot-Span Slot Nozzle with a Linearly Varying
Exit Profile

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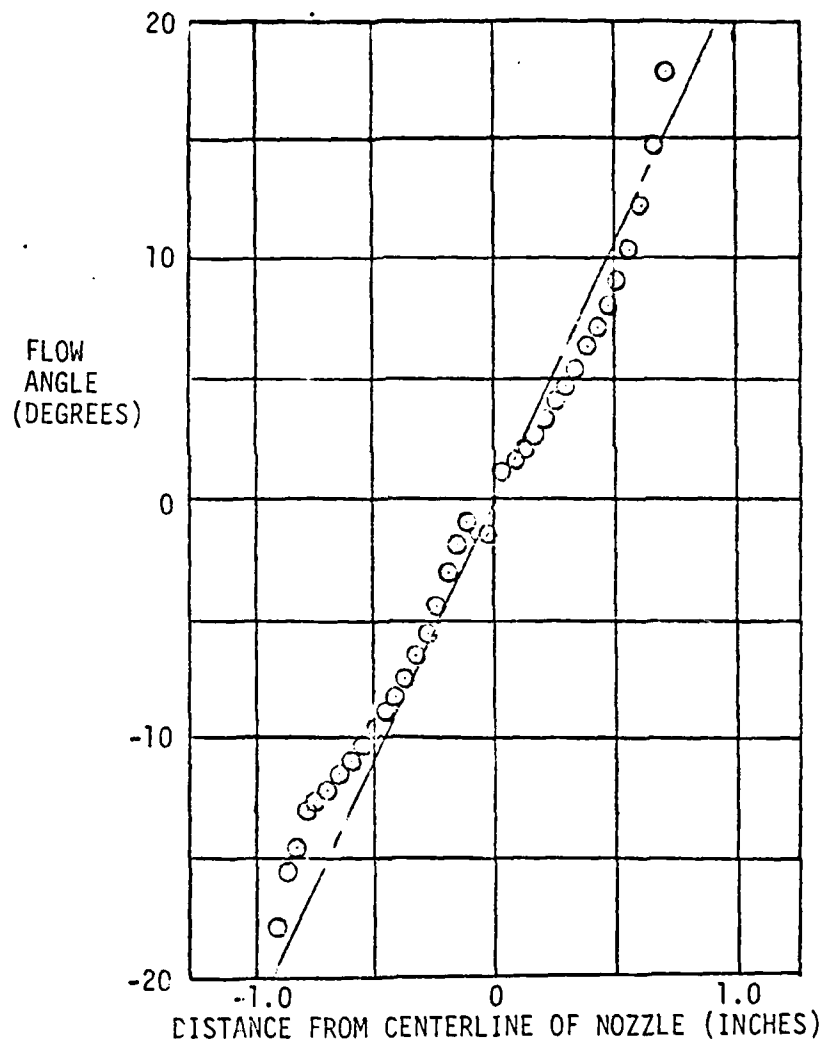
Figure 7-70 Measured Flow Angles for Linearly Varying Slot Nozzle
Configuration
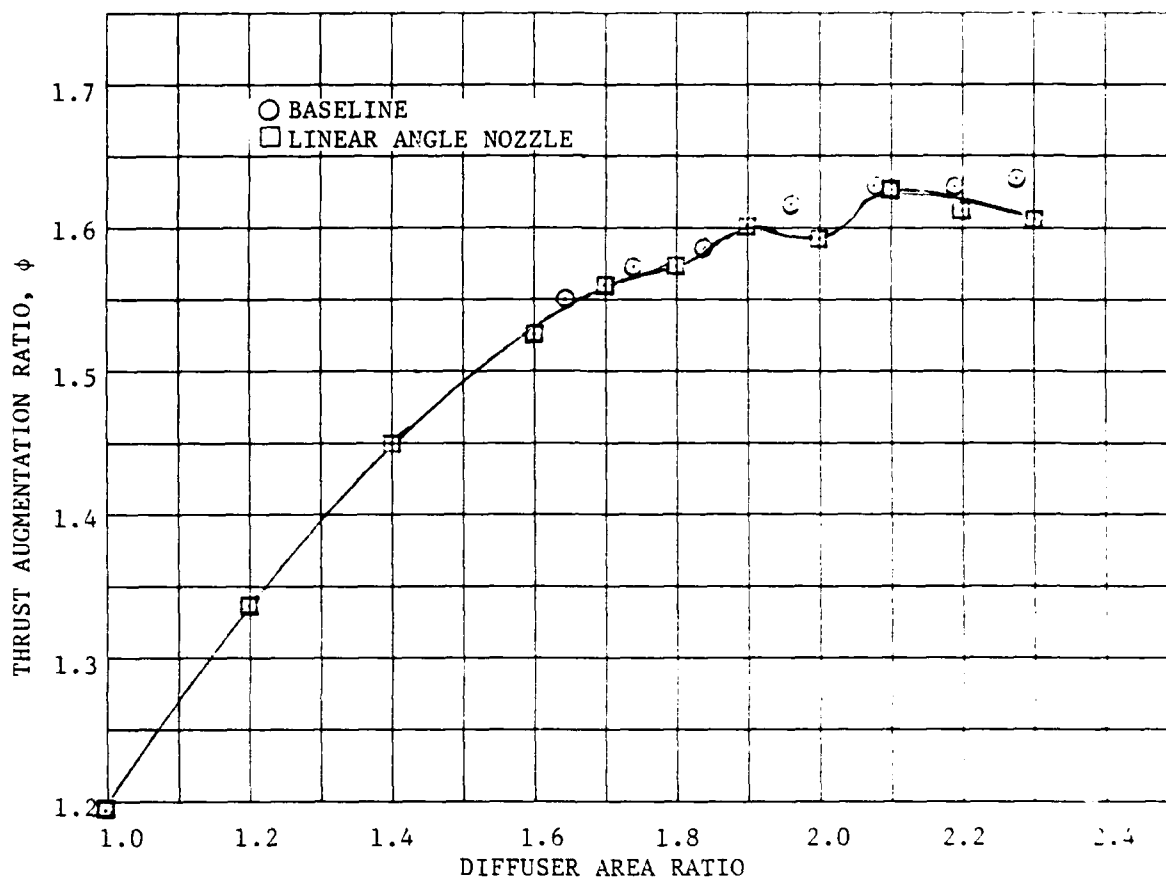
Figure 7-71 Effect of Linear Jet Angle Distribution on
Thrust Augmentation

7.5   SUMMARY

The improved version of the jet mixing program has been shown to be a
useful tool for analyzing the performance of centerbody nozzles. Test and
analysis have shown that the thrust augmentation of the XFV-12A wing can
be increased from 1.46 to 1.51 by increasing the hypermixing jet angle
from 7° to 22.5°. Similar development of an asymmetric centerbody
configuration resulted in a peak ϕ of 1.55. A symmetric cross slot
centerbody with optimized bowtie ratio, cross slot flow split and aspect
ratio, and cross slot exit flow angularly gave a peak   of 1.64.
Pressure distributions on the Coanda surfaces suggested that ϕ was limited
to this value by the appearance of separation bubbles. This effect should
be further studied.

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8.0 SHROUD STUDY

8.1 GENERAL

Ejector thrust augmentation involves no more than a novel application of the familiar circulation theorem of aerodynamic lift. An isolated jet induces an essentially lateral flow, as seen in Figure 8-1. The pressure and velocity of the secondary flow approaching an ejector is altered by the presence of the shroud. A circulation which redirects the secondary flow through the ejector is generated around each of the ejector flaps. The
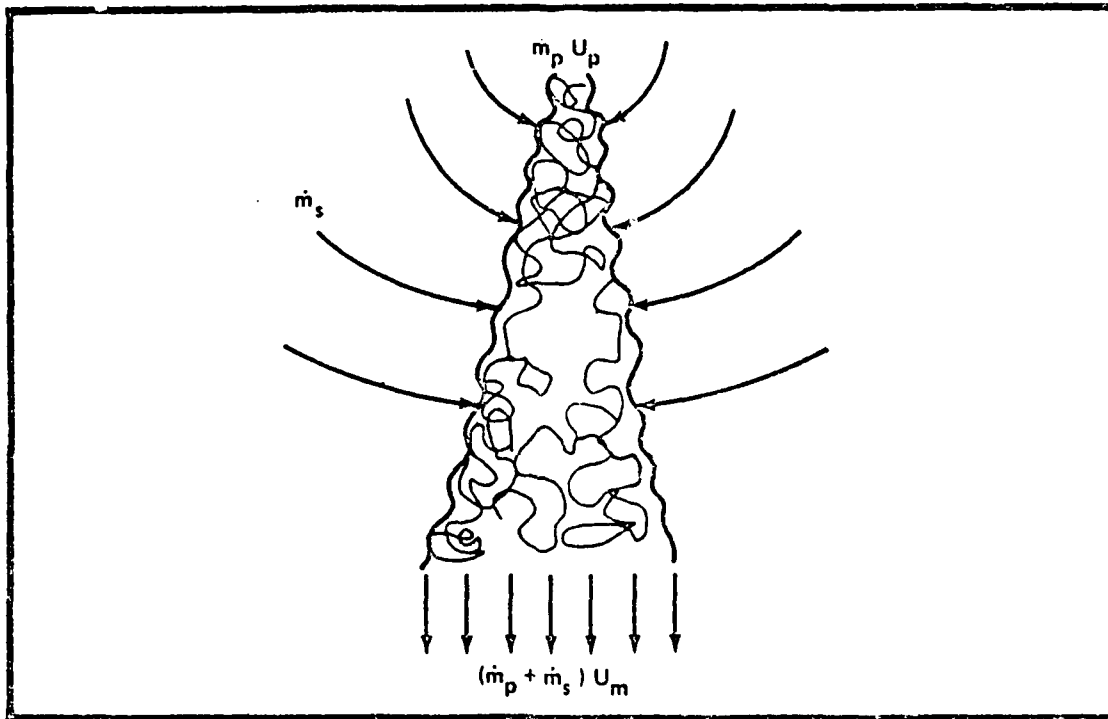
Figure 8-1 Streamlines of the Flow Induced by a Free Jet

flaps are thus considered to be "flying" in the entrained velocity field of the jet, Figure 8-2, and they experience a force related to the lift developed on a wing fixed in a moving stream. The thrust augmentation ratio can therefore be defined as the ratio of the primary jet thrust plus the axial force on the flaps to the isentropic thrust of the primary mass:

$$\phi = \frac{T + F}{\dot{m}V}$$

The theoretical augmentation can be increased by changing the shape and location of the shroud. But, in addition, the forces generated on the shroud due to the augmented thrust must be carried as a pressure loading around the shroud surface. Flow separation from the shroud will result in
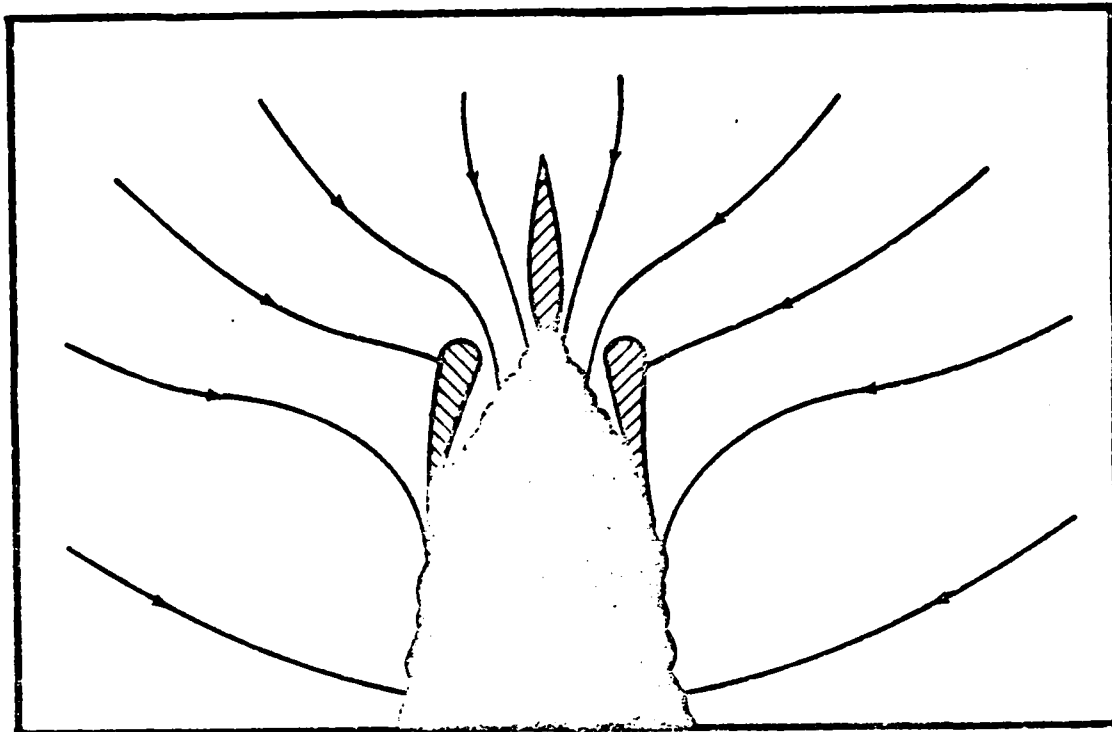
Figure 8-2 Entrainment by the Primary Jet Induces a Secondary Flow

a loss in performance for the augmenter. Therefore, it is important that consideration be given to understanding the shroud as an ejector component. This section presents experimental and analytical results of the effects of shroud related parameters such as flap length, flap asymmetries, flaperons, etc., on ejector performance.

8.2   EFFECT OF FLAP LENGTH

A series of experimental runs were conducted to determine the effect of flap length on ejector performance. The tests were conducted utilizing the .2 scale model on test rig no. 5. The series consisted of runs in which the length of the seven-inch straight flaps was increased to 9.5 inches and 12.3 inches, respectively. This resulted in L/W's of 1.5 (baseline), 2.0, and 2.5. The effect of increased flap lengths on thrust augmentation ($\emptyset$) is shown in Figure 8-3. It is seen that $\emptyset$ is independent of increased flap lengths for diffuser area ratios (DARs) less than 1.6. However, for DAR's greater than 1.6, it is seen that increasing flap length increases $\emptyset$ for a given DAR and increases the value of DAR at which maximum $\emptyset$ occurs. These resul   are consistent with those reported in Reference (30).

The effect of flap length on the distribution of the aft flap Coanda surface pressures is shown in Figure 8-4. In general, it is seen that

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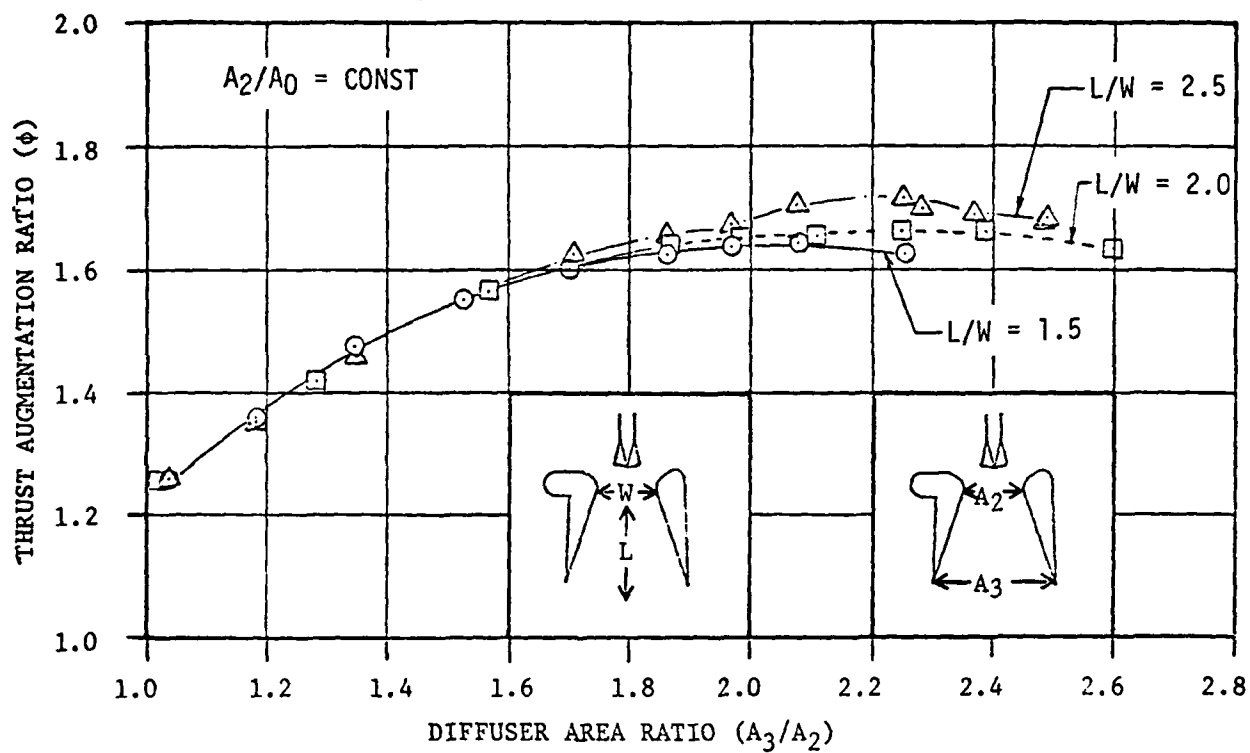
Figure 8-3   Effect of Flap Length on Thrust Augmentation Ratio

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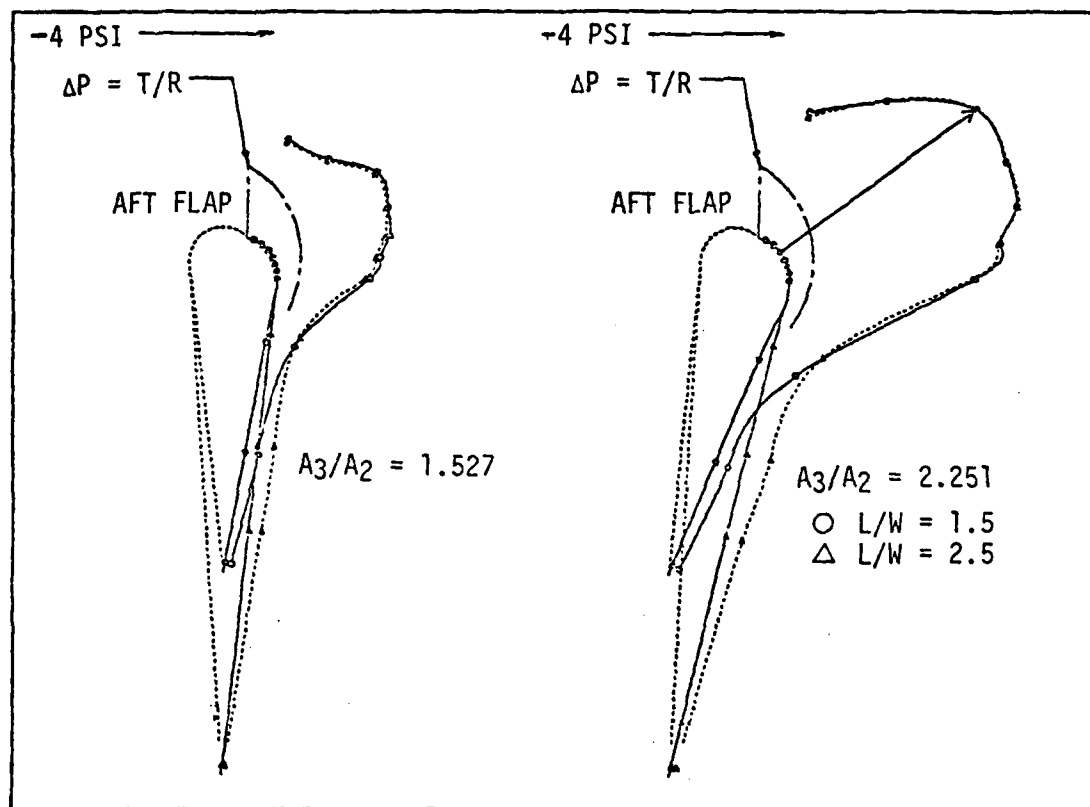
Figure 8-4 Effect of Flap Length on Surface Pressure Distribution

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increased flap length has little effect on either the magnitude of
pressure or its distribution around the Coanda surfaces. The same trend
was observed for the forward flap.

In order to increase understanding of flap length effects, augmenter exit
surveys were obtained at diffuser area ratios of 1.183 and 1.965. The
data from these surveys were integrated to obtain momentum, mass flow,
thrust and a velocity profile factor (β). The effect of flap length on
exit mass flow in presented in Figure 8-5. The velocity profile factor is
shown in Figure 8-6.

The independence of ϕ with flap length for area ratios less than 1.6 is
due to at least two cancelling effects. First, for a given DAR, increased
flap length would result in reduced flap angle, i.e., the flaps are flying
at a lower angle of attack. Therefore, the velocities induced by the
flaps at the throat are lower. However, the throat velocity data indicate
no effect due to flap length. Hence, it is likely that increased mixing
cancels the effect of angle.

At diffuser area ratios greater than 1.6, the baseline configuration is
stalled. Stall is a condition in which the resultant force continues to
increase, but the force vector for each shroud is tilted downward. A
sketch of this condition is shown in Figure 8-7. When augmentation is
limited by this phenomenon, the secondary mass flow, i.e., throat
velocity, continues to increase past the point of maximum thrust.

At large diffuser area ratios, the increases in ϕ at constant diffuser
area ratio with increased L/W is due to a reduced flap angle which tilts
the resultant force vector upwards. Secondly, the jets are brought closer
together resulting in a lower velocity profile shape factor. This latter
result is seen in Figure 8-6

8.3   FLAP LENGTH ASYMMETRIES

The integration of an ejector into the wing of an aircraft requires
compromises in both wing and ejector geometry. The XFV-12A required
tapered ejectors. However, design studies showed that a nearly
rectangular ejector could be designed for the XFV-12C configuration. Even
though the ejector could be nearly rectangular, it would require
asymmetries in flap length. Therefore, a test program was initiated to
assess the effects of different fore and aft flap lengths on ejector
performance. Unfortunately, only one asymmetric flap length test series
was completed prior to program termination.

The results of a test in which the aft flap length was 1.78 inches longer
(model scale) than the forward flap is shown in Figure 8-8. The DAR for
the asymmetric flap configuration has been defined in two different ways
and the data have been presented accordingly. It is seen that a 25 per
cent increase in aft flap length had no effect on peak ϕ. Furthermore,
using the symmetric definition of exit area yielded a better correlation
than using the asymmetric definition. The data obtained for this series
also indicated no effect of flap length asymmetries on thrust angle.

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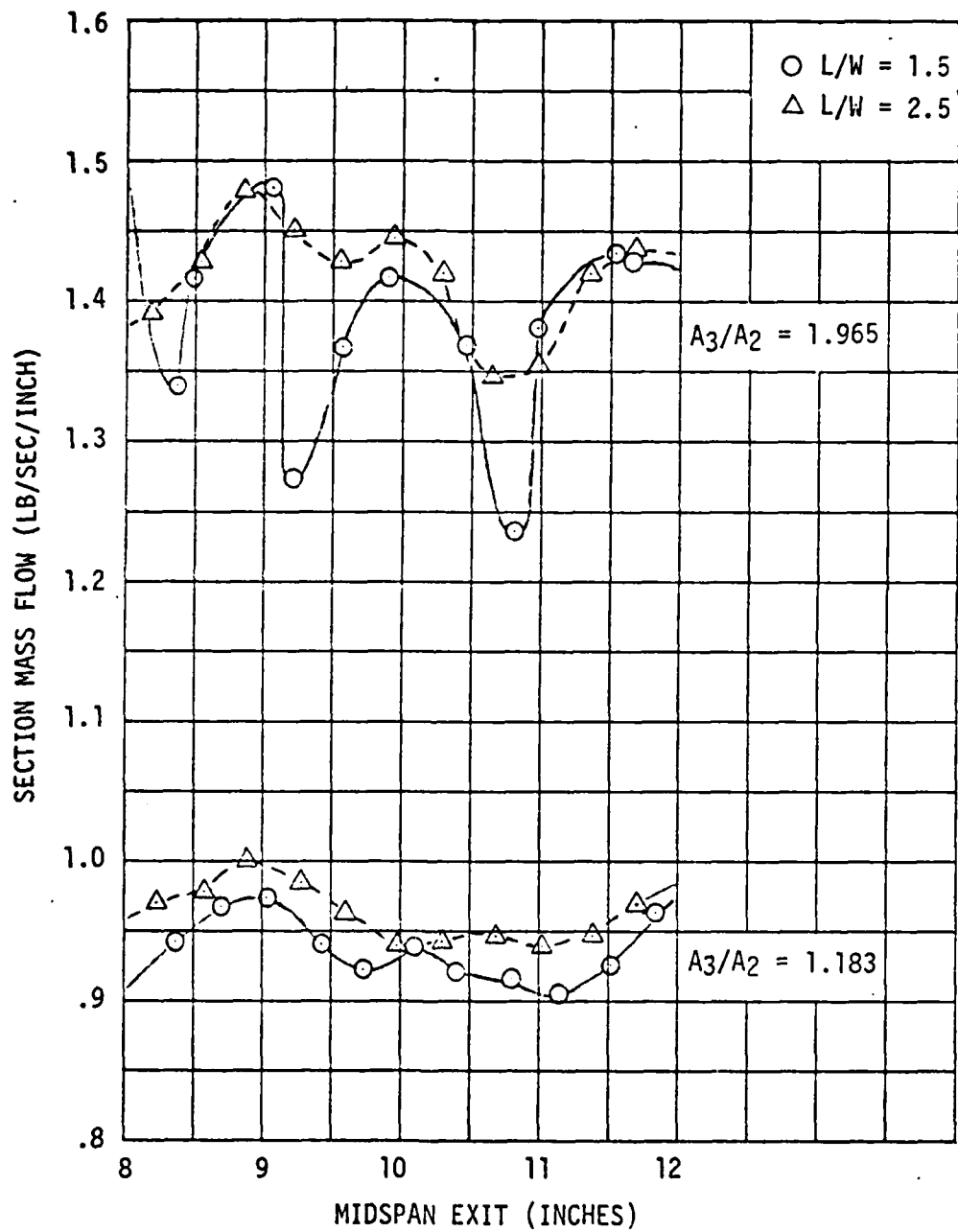
Figure 8-5   Effect of Flap Length on Section Mass Flow

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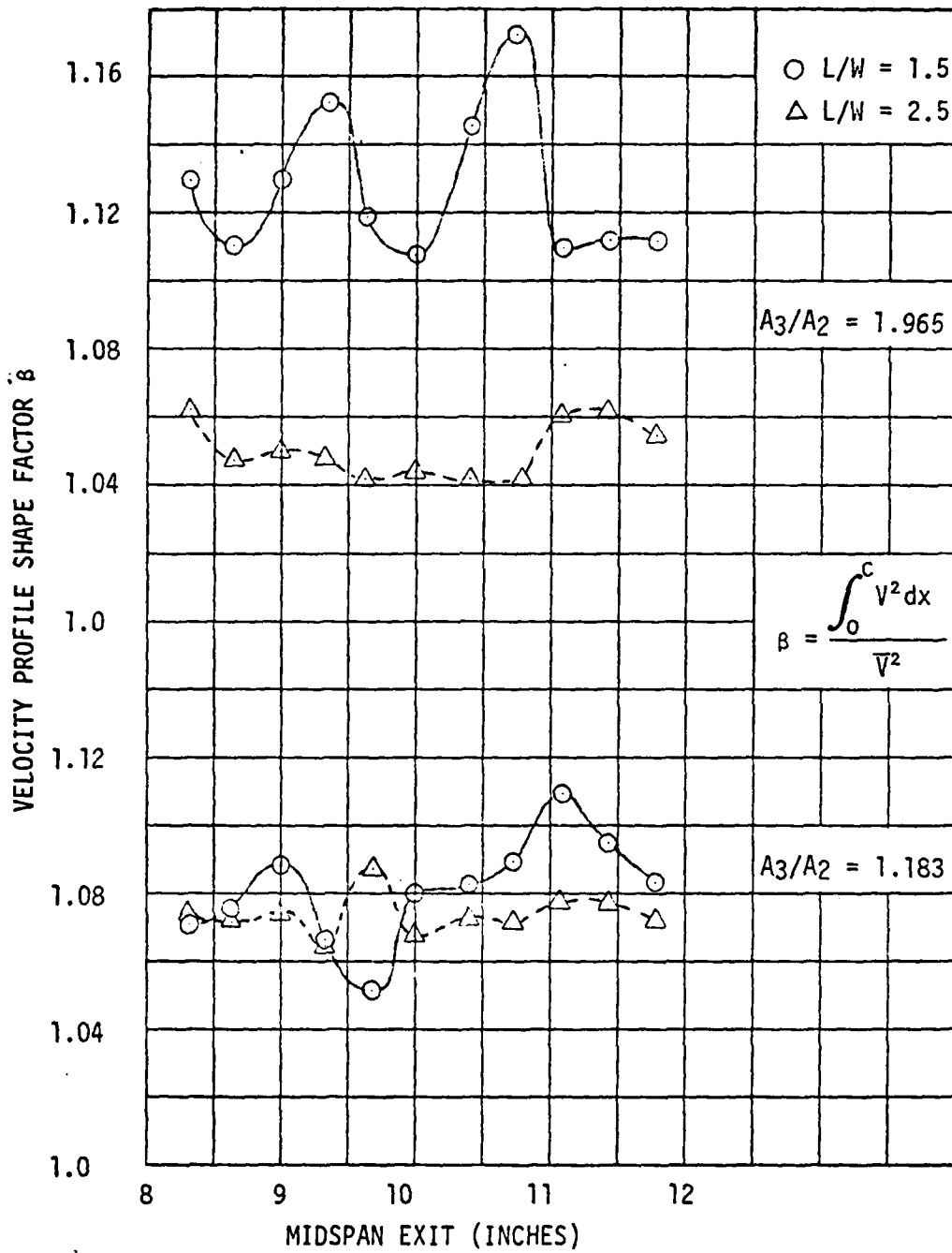
Figure 8-6 Effect of Flap Length On Velocity Profile Shape Factor

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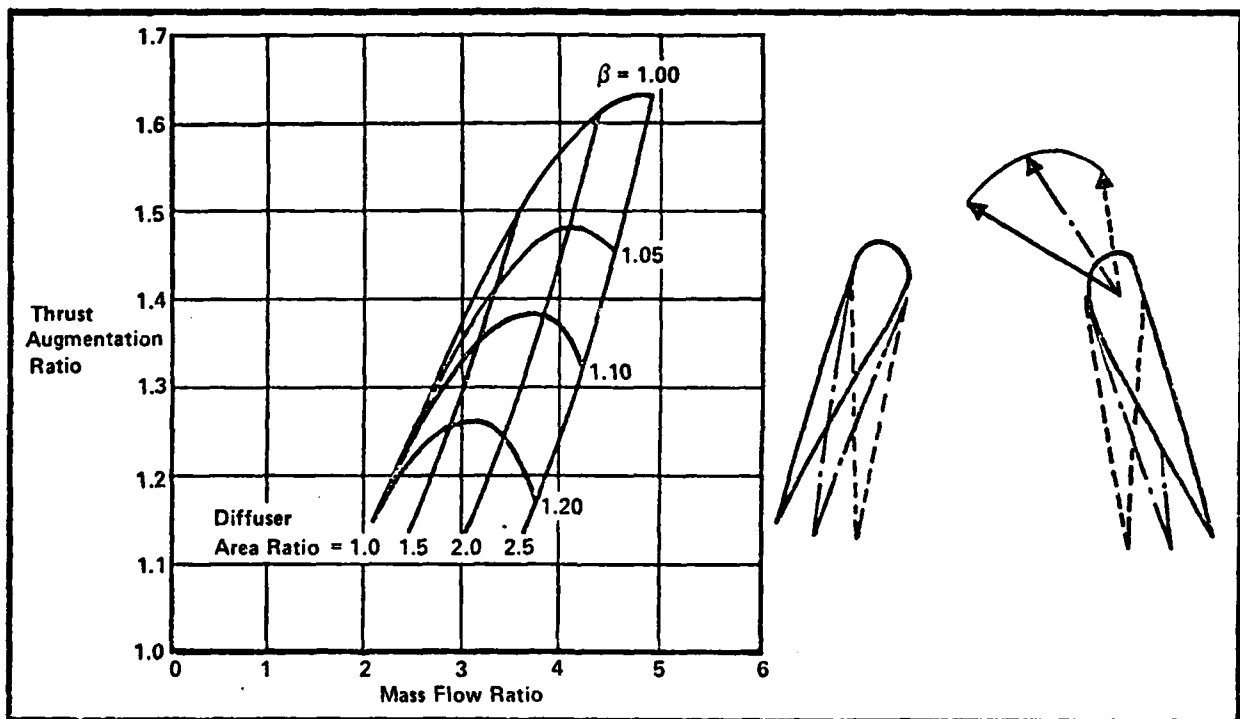
Figure 8-7   Inclination of the Thrust Vector Limits the Thrust Augmentation

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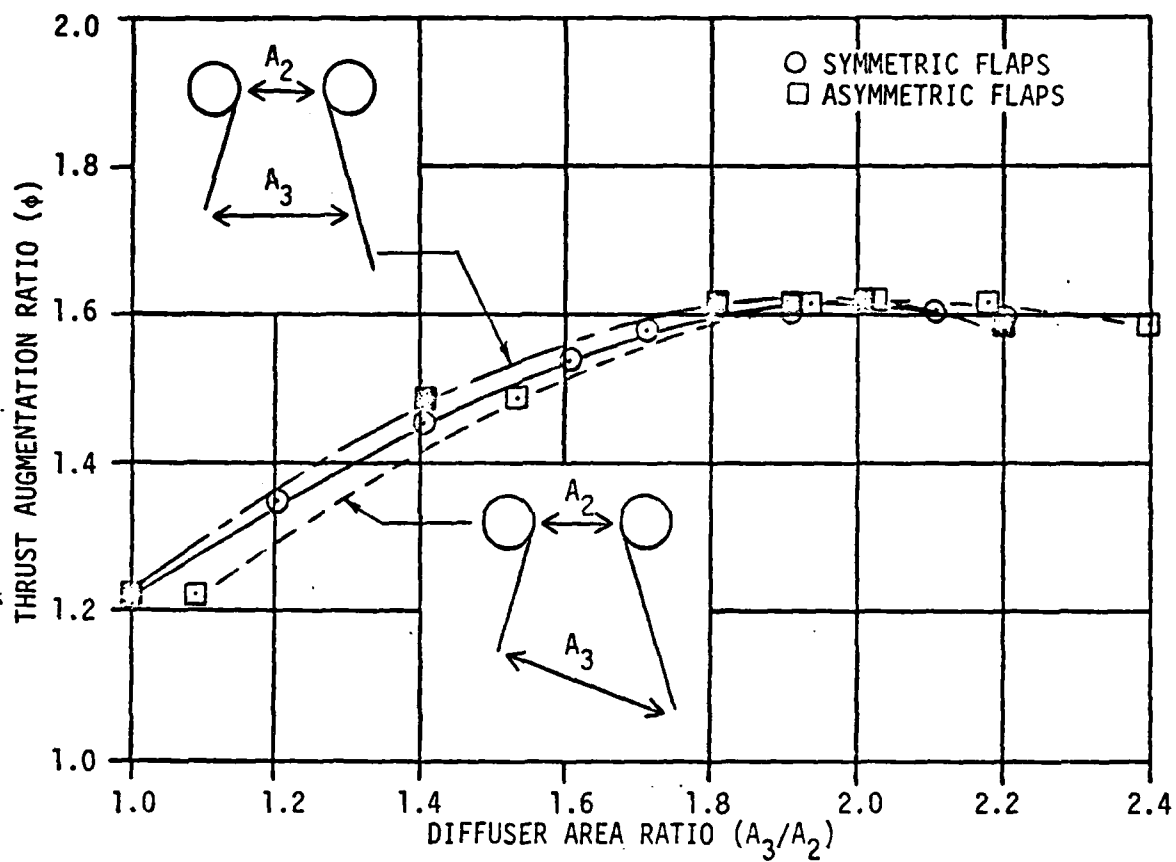
Figure 8-8 Effect of Asymmetric Flap Length on ϕ

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8.4 LOCAL SEPARATION EFFECTS (BUBBLES)

Careful study of the thrust augmentation (ϕ) variation with diffuser area ratio (DAR) shown in Figure 8-9 shows a characteristic that has been typical for all of the high performing symmetric cross-slot (CS) nozzles. Namely, a break in ϕ between DAR's of 1.6 and 1.8. This characteristic is thought to be due to a local separation that first occurs on the forward flap Coanda surface and then, at a slightly higher DAR, on the aft flap.
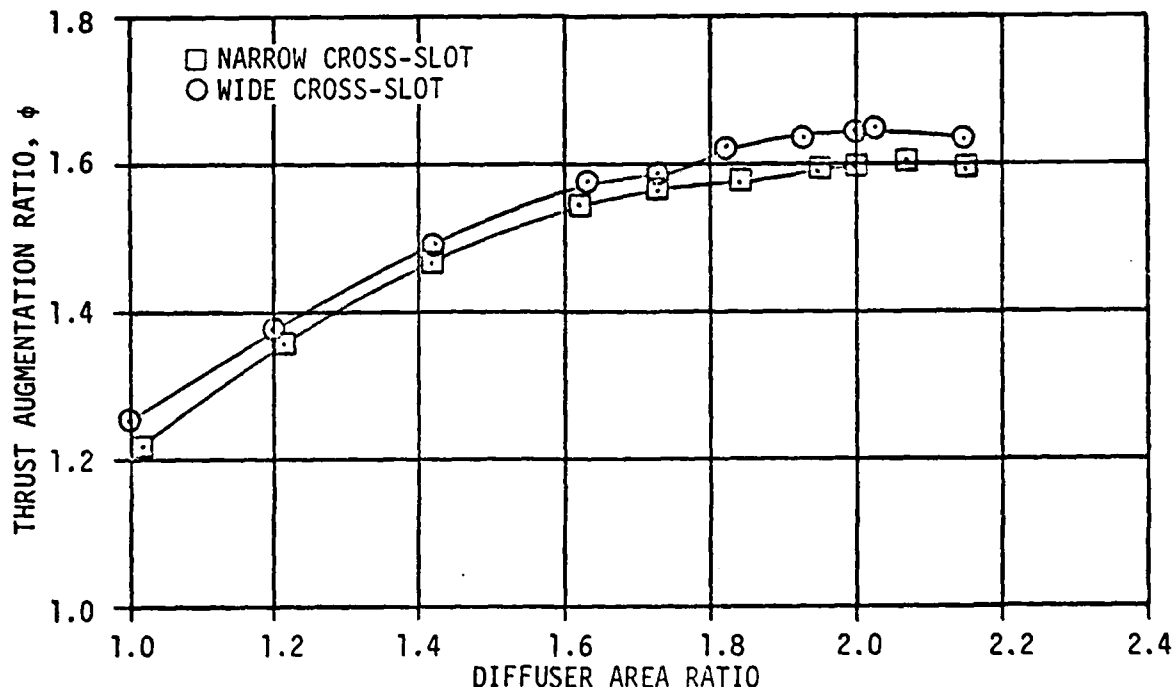
Figure 8-9 Effect of Cross-Slot Width on Thrust Augmentation Ratio

Figure 8-10 presents a comparison of the pressure distribution on the forward flap surface for both narrow and wide cross-slot nozzles. Figures 8-10a and 8-10b show that both nozzles produce the same pressure distributions, however, the wide CS nozzle produces a more negative value of pressure. The irregularities in pressure occurring at about 20 nozzle thicknesses (s/t) around the surfaces are thought to be due to local separation. The possibility of expansion and compression waves has been ruled out because of the relatively large distance from the nozzle.

This Coanda surface pressure data has been replotted in Figure 8-11 to show the variation of surface pressure with diffuser area ratio for a given pressure tap location. At the fourth tap (s/t $\approx$ 20) the previously discussed irregularities can be clearly seen.

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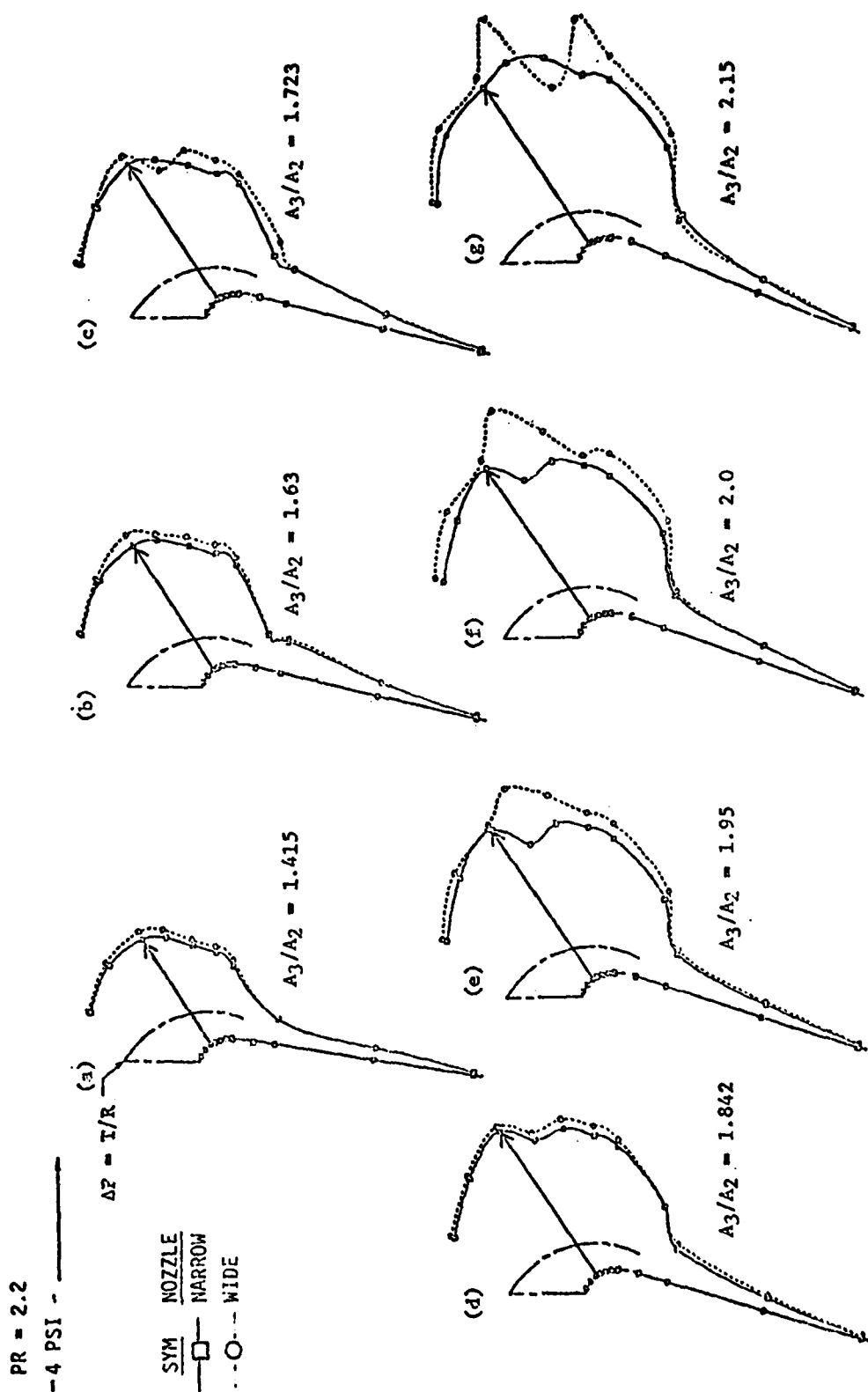
Figure 8-10 Forward Flap Surface Pressures for Narrow and Wide Nozzles
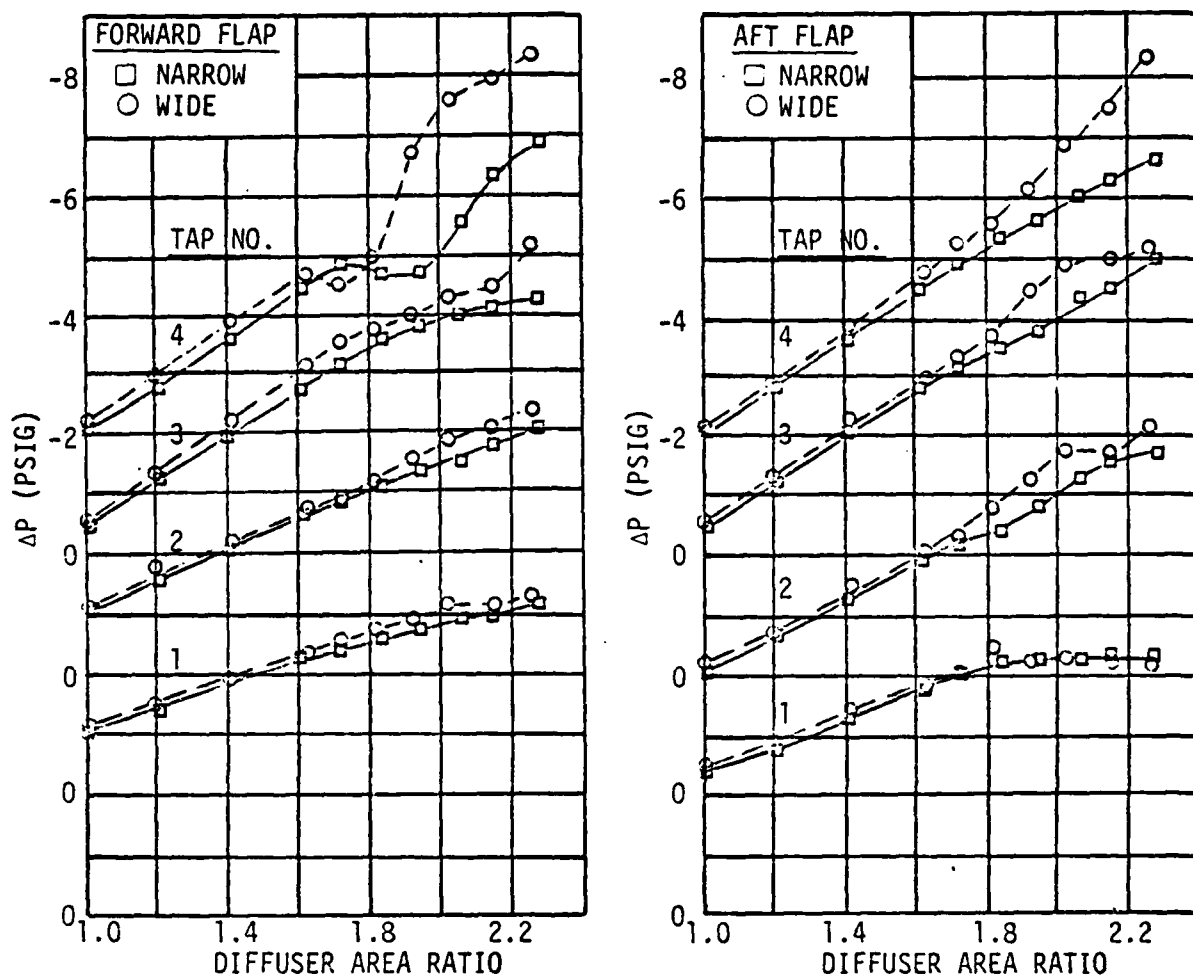
Figure 8-11 Variation of Surface Pressure with Diffuser Area Ratio

Additional data which show the possibility of local separation on the Coanda surfaces is shown in Figures 8-12. These data show the effect of artificial surface roughness (grit) on surface pressure. The pressures are plotted versus arc length (measured from nozzle exit). The grit was placed between pressure taps three and four. Figure 8-12 shows that without grit the aft flap has a very smooth pressure distribution. However, the addition of grit creates a local separation with a pressure loss downstream of the separation point. The pressure distributions on the forward flap are very similar with and without grit, i.e., local separation is indicated. The character of the aft flap pressure distribution with grit and the forward flap pressure distribution without grit are similar. The effect of grit on ejector performance is shown to be a loss in ϕ of at least .04.

Rockwell International
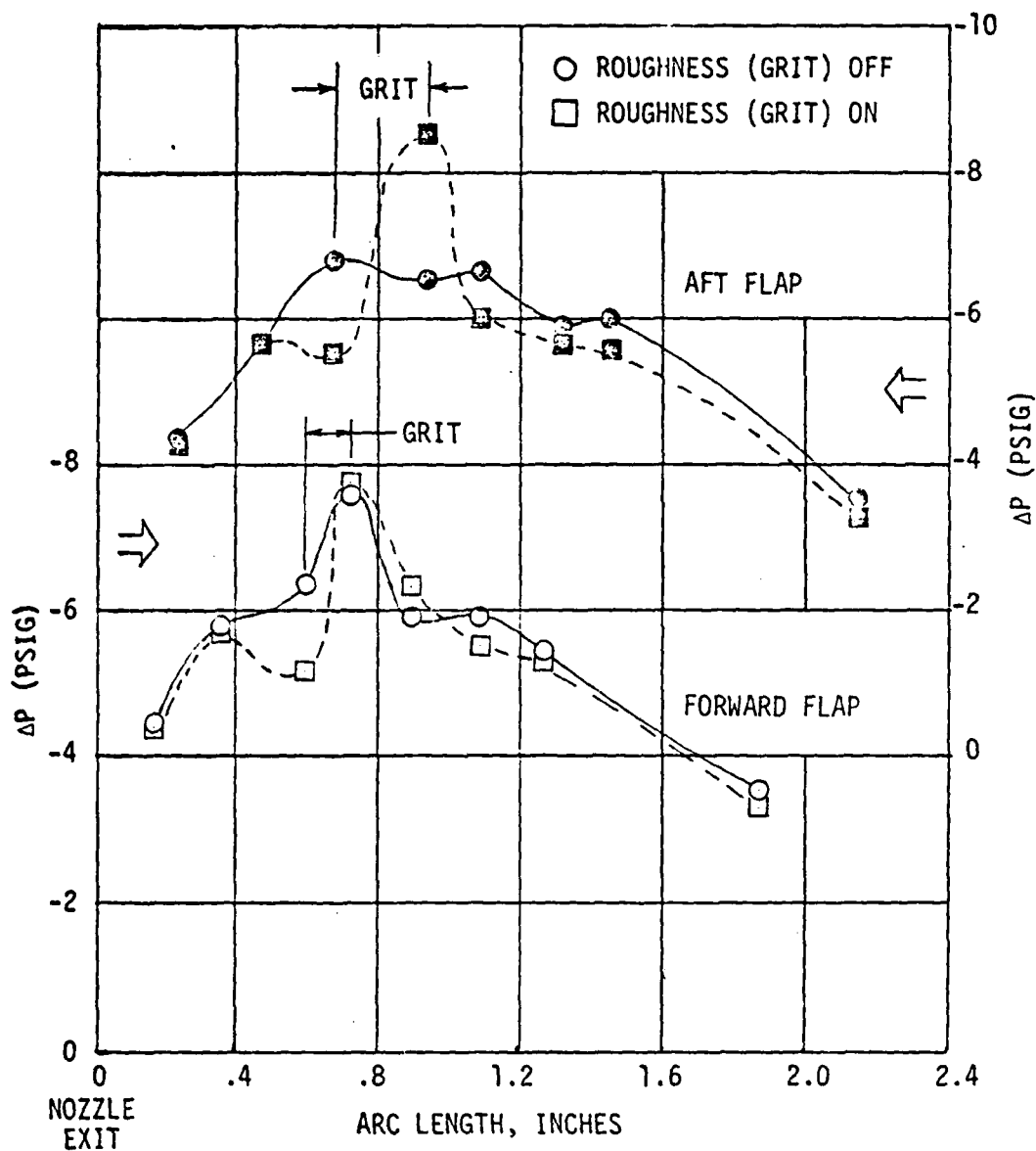
Figure 8-12 Effect of Roughness On Coanda Surface Pressures

It is concluded, based on these data, that both the forward and aft flaps experience local separation at s/t ≈ 20. The indicated separation location is near the point at which the pressure gradient becomes adverse.

8.5   INVISCID ANALYSIS OF SHROUD

An analysis of the shroud has been conducted using an inviscid computer code. The procedure employed was to set the secondary flow rate through the shroud by adjusting the strengths of an assumed center jet sink

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distribution. The strengths were adjusted so that the throat velocity obtained from the inviscid code was equal to a desired value (obtained from experiment or the jet mixing analysis).

Figure 8-13 shows the modeling used to represent the .2 scale rectangular wing at a diffuser area ratio of 2.0 The computed streamlines are also shown. Figures 8-14 and 8-15 present a comparison of experimental and computed velocities at the ejector throat and at the inlet plane. Figure 8-16 shows a comparison of chordwise flow angles at the inlet plane. Each figure presents experimental data at a span station under a cross-slot nozzle and one under the span slot. The secondary flow is seen to be almost two-dimensional. The agreement between test and theory is excellent and shows that the inviscid code is quite useful in determining inlet flow characteristics.

A comparison of computed and experimental surface pressures are shown in Figure 8-17. The pressure due to Coanda jet turning, i.e., $\Delta P =$ Thrust/Radius, has been removed from the experimental data. The throat velocities obtained from the jet mixing code were used as control points.

Interestingly, the lowest pressures do not occur at the throat. It is concluded that the shroud surface pressures are composed of a part due to channel-type flow and a part due to airfoil-type flow. The superposition of the two types moves the minimum pressure away from the throat and towards the shroud leading edge.

The excellent agreement of the inviscid code with experimental data led to a request to increase the number of pressure taps on the shroud surfaces. However, detail pressures were obtained for only one centerbody nozzle configuration prior to XFV-12A program termination. A comparison of theory and experiment for that configuration is shown in Figure 8-18. The comparisons shown have the largest discrepancy at the "leading edge." The potential flow solution indicates a large adverse pressure gradient just upstream of the nozzle. It is possible that the flow could be separated in this region and be re-entrained further downstream. An analogy can be drawn between this "postulated separation" and typical airfoil leading edge separation. In the latter, leading edge separation results in a rapid increase in drag with no appreciable change in lift. Separation of this type for the ejector would be quite detrimental to performance and difficult to detect without further detailed instrumentation. Further study of ejector leading edge separation was planned.
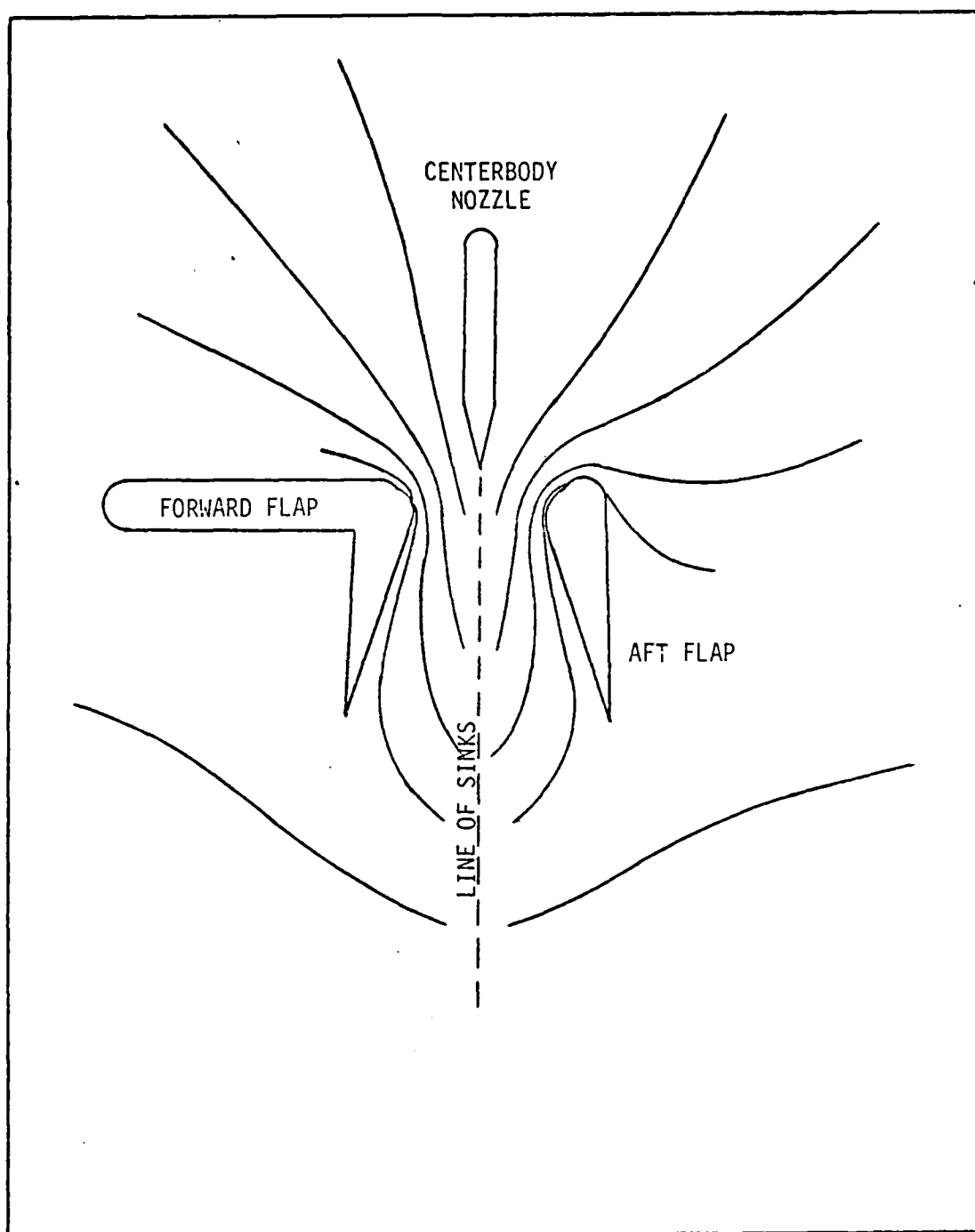
Figure 8-13 Entrainment by the Primary Jet Induces A Secondary Flow
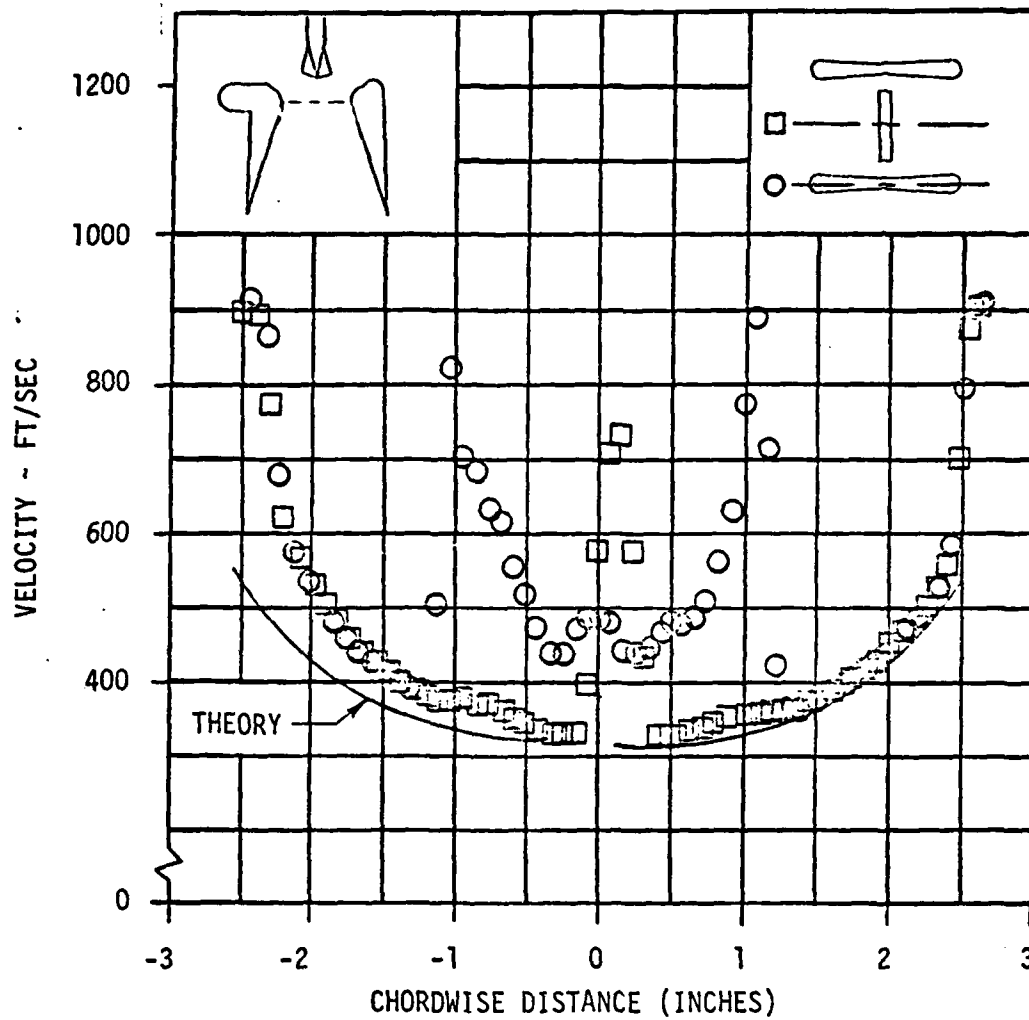
Figure 8-14 Comparison of Experimental and Theoretical Velocities at the Throat Plane

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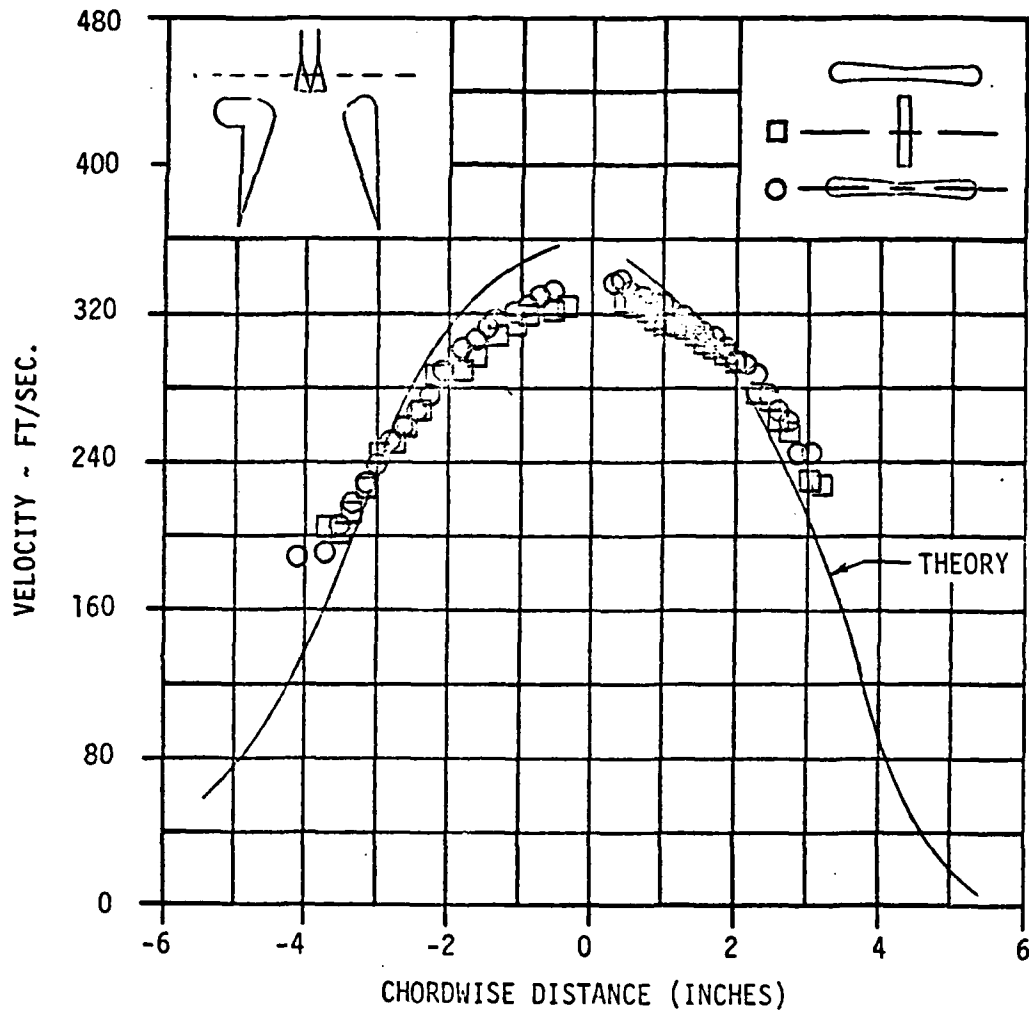
Figure 8-15 Comparison of Experimental and Theoretical Velocities At A
Nozzle Plane

Rockwell International
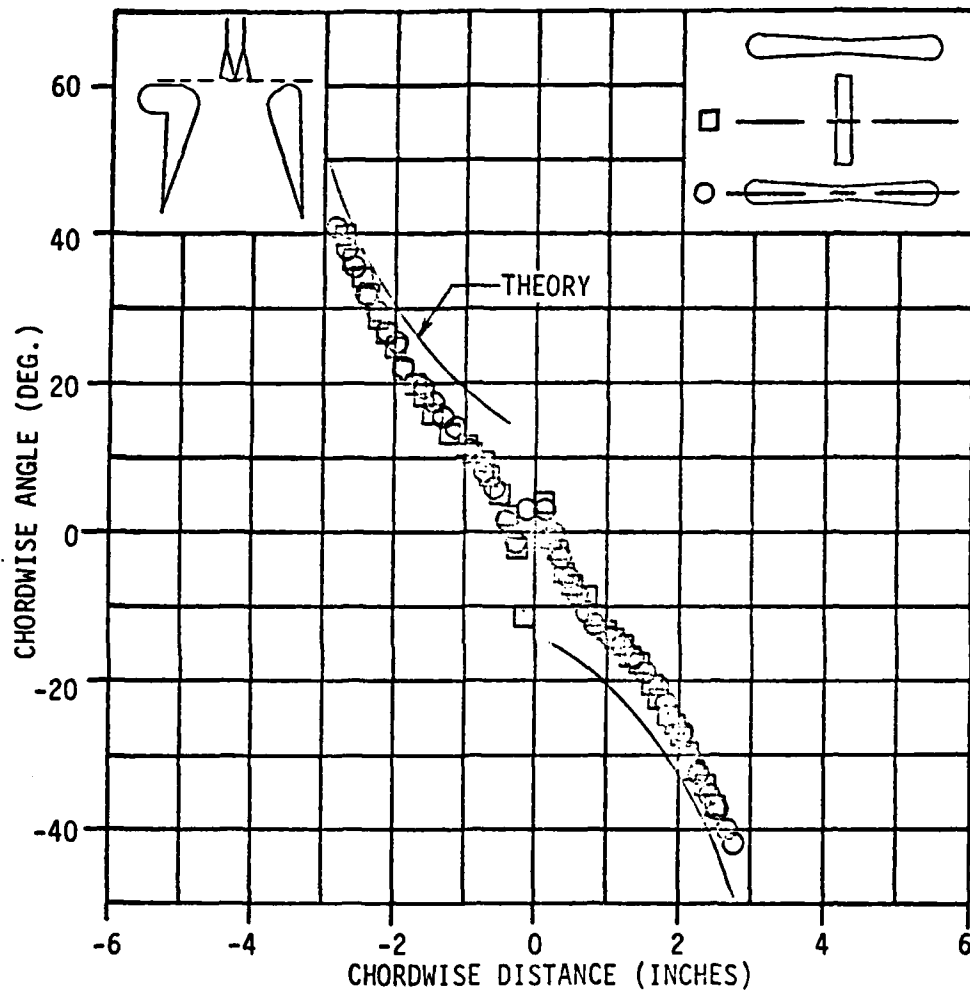
Figure 8-16 Comparison of Experimental and Theoretical
Flow Angles At the Inlet Plane

Rockwell International
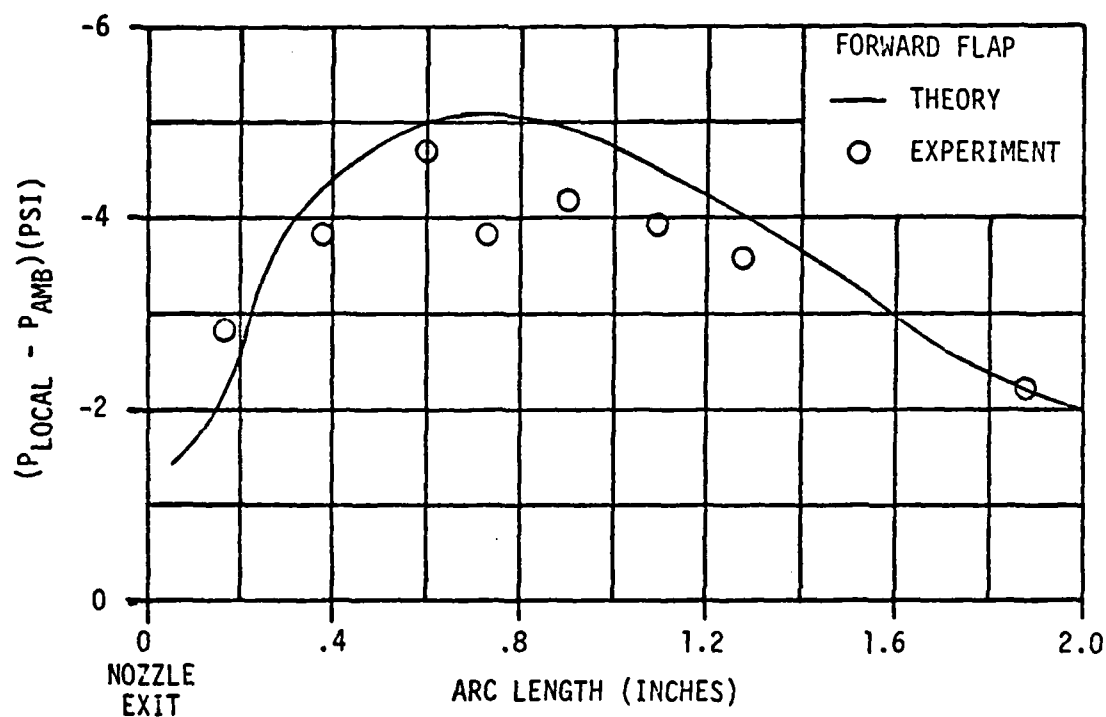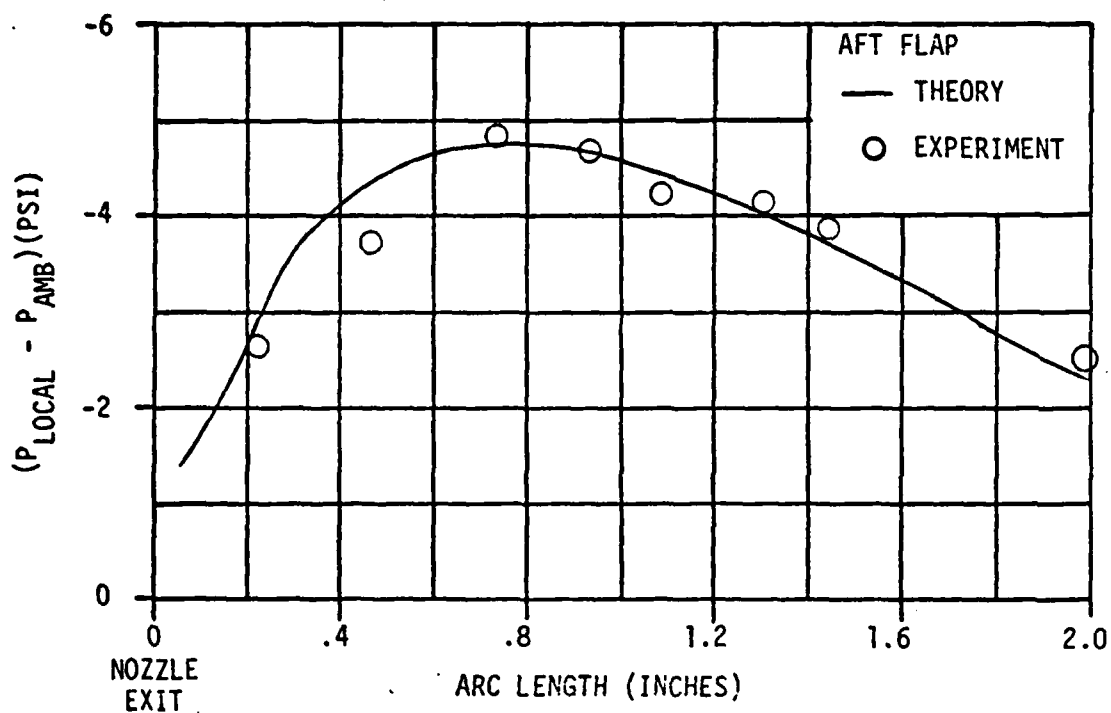
Figure 8-17 Comparison of Experimental and Theoretical Surface Pressures

Rockwell International
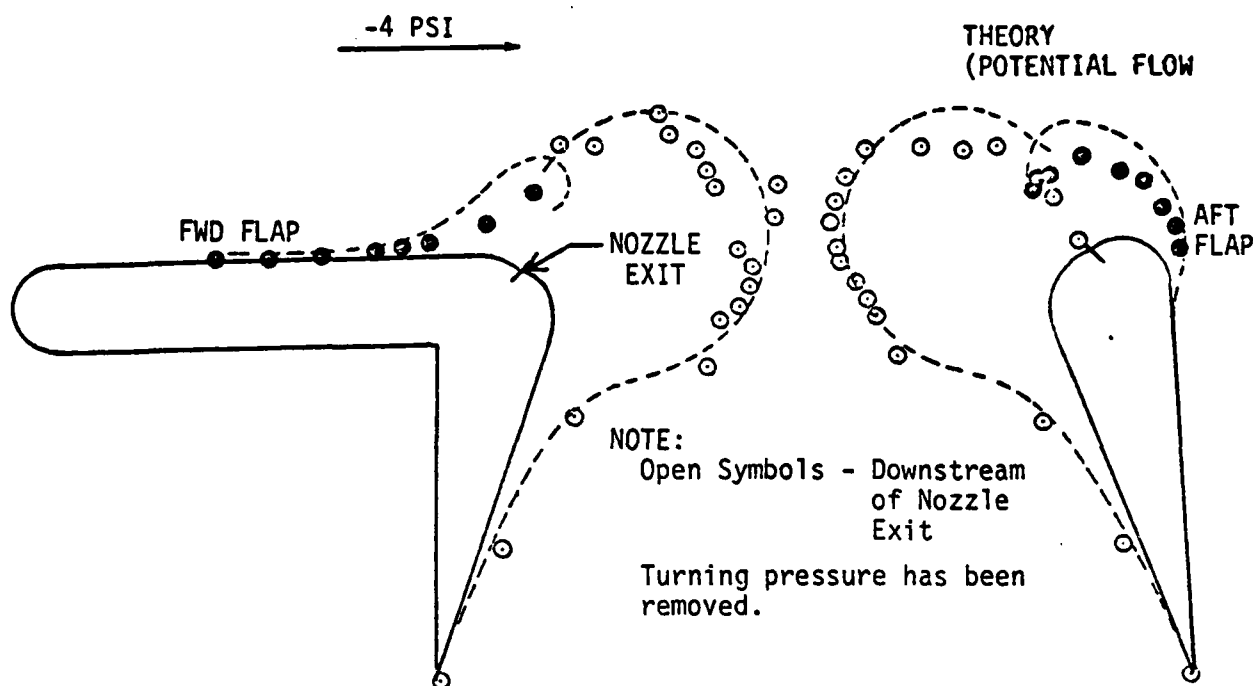
Figure 8-18   Comparison of Predicted and Experimental Shroud Pressure
Distributions

8.6   FLAPERONS

Ejector performance can be increased through improved nozzle designs
(mixing) or through improved shroud geometry. The use of flap trailing
edge tabs or flaperons as a means of improving external diffusion has been
reported in Reference (31). The flaperons have an effect similar to that
of trailing edge flaps with blowing. First, there is an increase in lift
due to trailing edge camber and, secondly, lift is increased as a result
of the jet flap diffusion reported in Reference (32). Therefore, a series
of tests were conducted on the 0.20 rectangular ejector to study flaperon
effectiveness for the XFV-12A shroud geometry.

A sketch of the flaperon configuration tested is shown in Figure 8-19. A
comparison of thrust augmentation and throat velocity for the straight
flap and flaperon configurations are shown in Figure 8-20 and 8-21,
respectively. The results show an average increase in thrust augmentation
of .04 for diffuser area ratios less than 1.8. However, there is no
increase in peak augmentation. The throat velocity continues to increase
past peak ϕ for both configurations. The flaperon configuration was
visually observed to have corner separation at the largest diffuser area
ratio.

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7" STRT FLAP (356-01042-063)
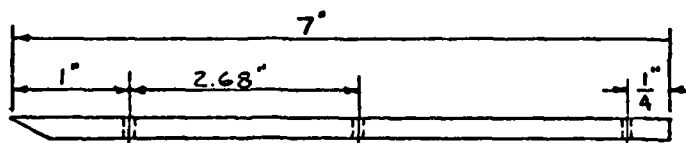
15° 7" FLAPERON (356-01042-181)
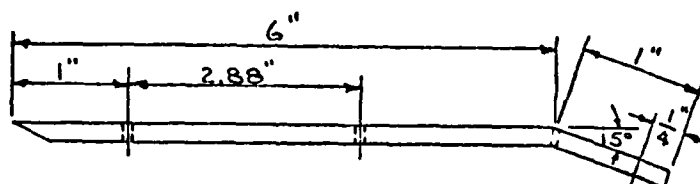
Figure 8-19   Sketch of the Flaperon Configuration

The lack of exit flow surveys and detailed pressure distributions make it difficult to determine why the flaperon configuration did not obtain a larger peak ϕ. It is postulated that two separate factors contribute to the poor peak performance. First, the previously discussed leading edge separation would occur at a lower diffuser area ratio with flaperons. Secondly, the data presented in Reference (31) showed a degradation in peak ϕ for a kinked flaperon versus a rounded tab.

Exit surveys and detailed leading edge pressure instrumentation would be desirable to develop a fuller understanding of flaperons.

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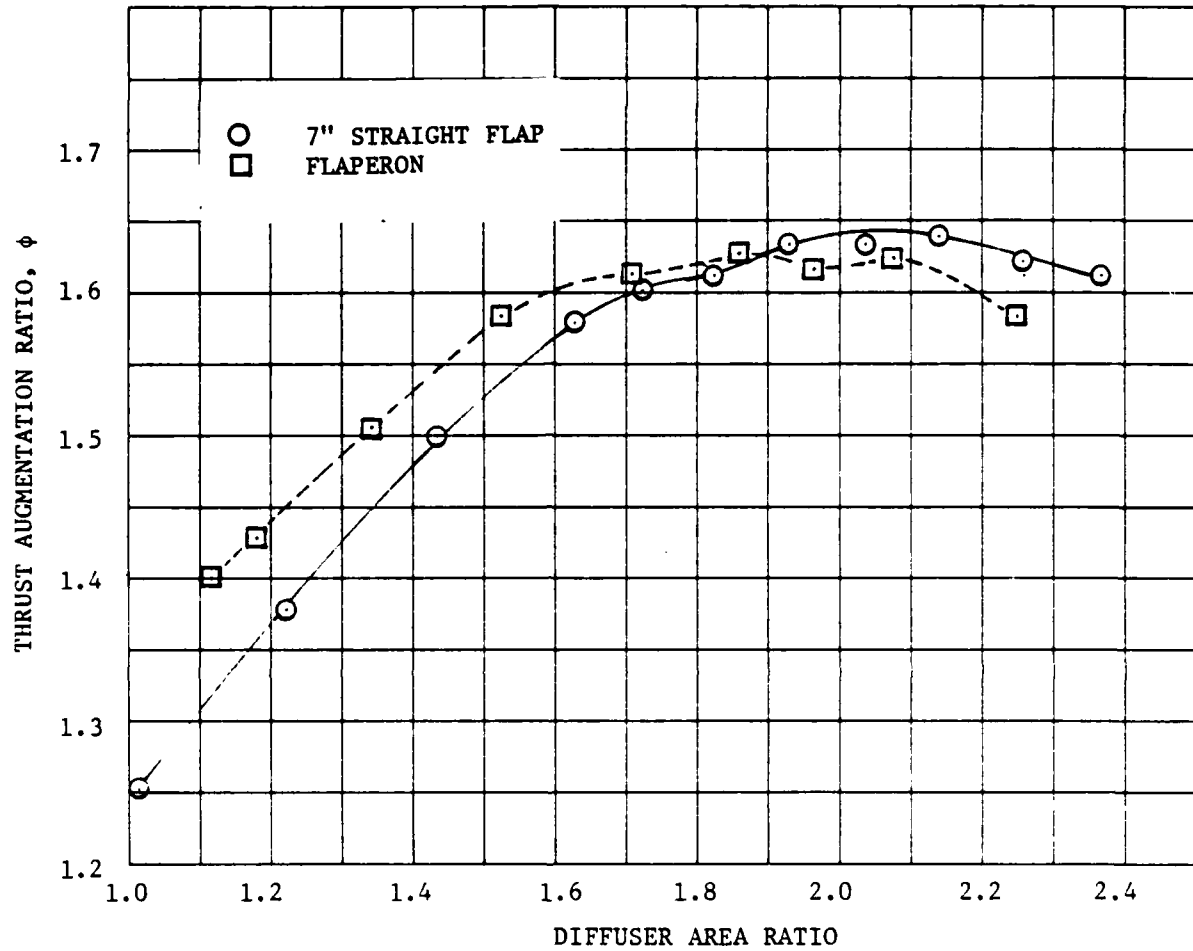
Figure 8-20 Comparison of Thrust Augmentation for the Straight
Flap and Flaperon Configuration

Rockwell International
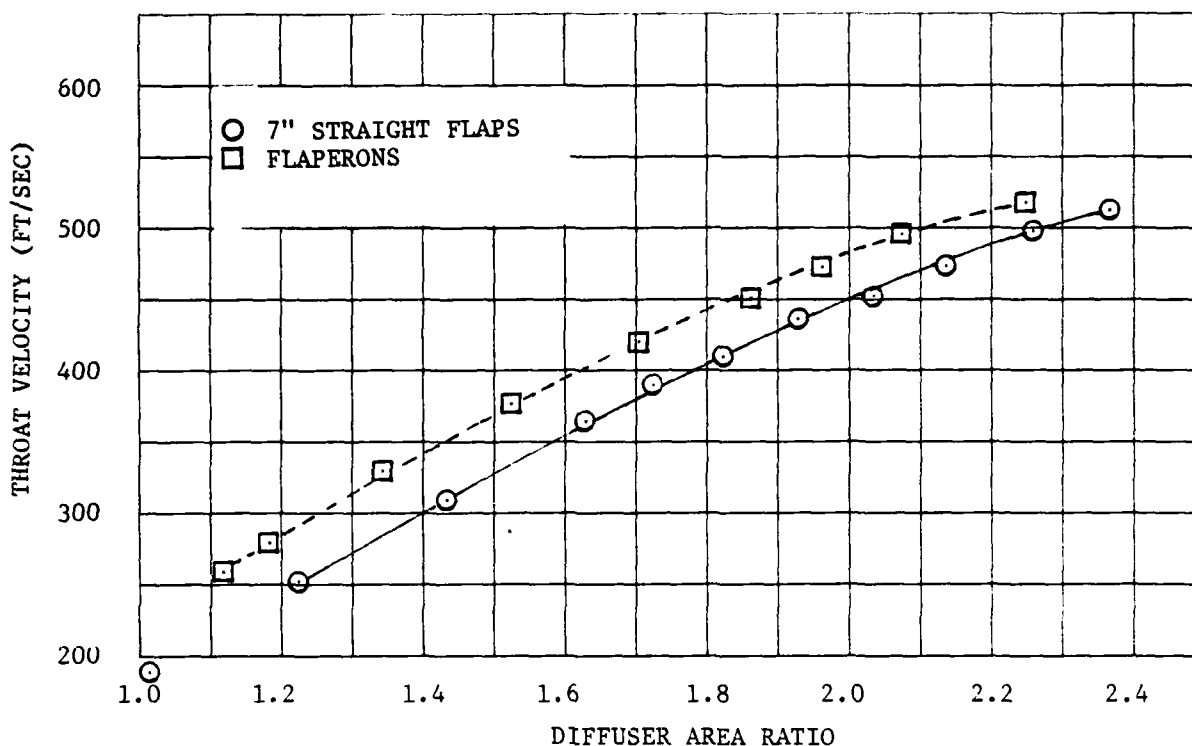
Figure 8-21 Comparison of Throat Velocity for the Straight
Flap and Flaperon Configuration

8.7   BOATTAIL ANGLE EFFECTS

Increases in the boattail angle of the centerbody nozzle result in
increased internal plenum area and a possible reduction in internal flow
losses. However, external flow separation or reduced inlet efficiency may
result. Therefore, an experimental investigation was conducted to
determine the effect of centerbody nozzle boattail angle on ejector
performance.

Three scale model centerbodies with boattail angles of 30°, 40°, and
50° were built and tested. The centerbodies were identical except for
boattail angle. The results of the test are shown in Figure 8-22 as
thrust augmentation versus diffuser area ratio. It can be seen that
boattail angle in the range tested had no effect on thrust augmentation.
Therefore, an improvement in full-scale plenum efficiency could be
obtained with increased boattail angle without a performance reduction.

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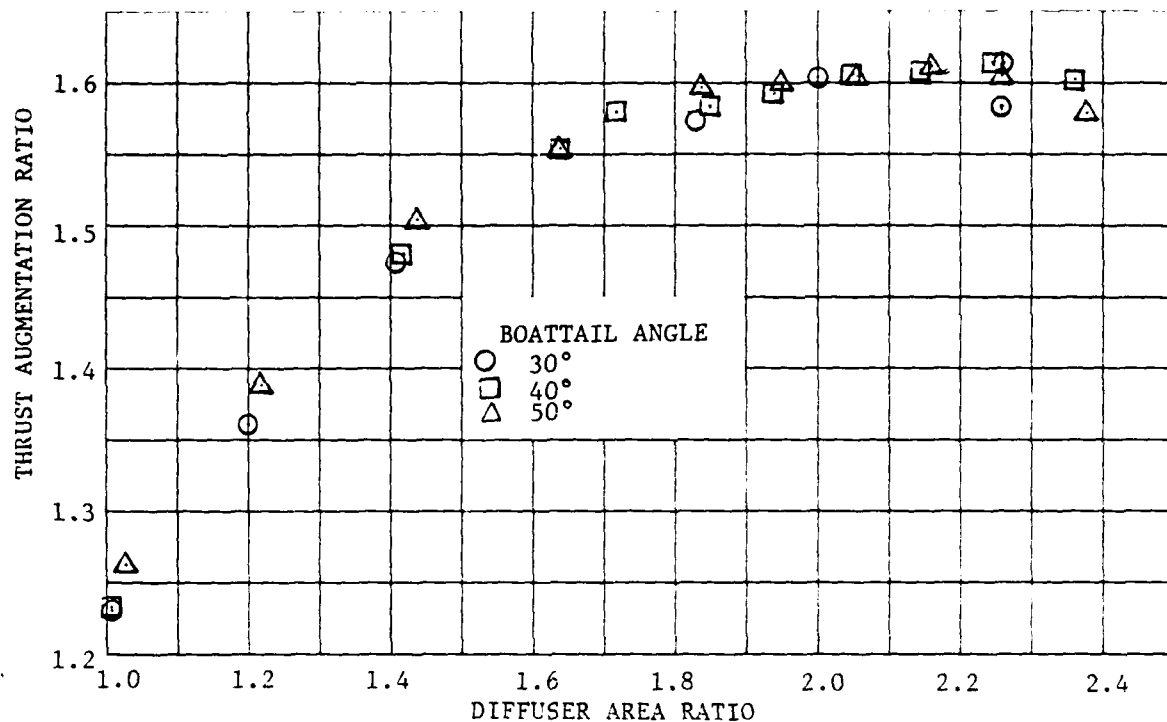
Figure 8-22   Effect of Boattail Angle on Thrust Augmentation

8.8   SUMMARY

An experimental and analytical study of shroud parameter effects on
ejector performance show the following: improvement in augmentation ratio
with longer shrouds are the result of more effective diffuser mixing;
modest asymmetries of ejector flap lengths do not degrade augmentation
ratio; local areas of separation and reattachment have been identified for
a XFV-12A type ejector; and flaperons are an effective method of
increasing shroud circulation lift until onset of separation. In
addition, a potential flow solution to predict shroud surface pressures
has been developed and gives results that compare favorably with
experimental data.

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9.0 FULL SCALE COMPONENTS

9.1 INTRODUCTION

Results of the Langley hover testing of the XFV-12 indicated that the lack of expected performance of the augmenters could be traceable to deficiencies in individual augmenter components, that is, diffuser, elevon, centerbody, and endwall blowers. An evaluation and improvement program was inaugurated to identify the deficiencies of each component and incorporate whatever improvements would be considered necessary or beneficial. The improvements to be made would come from the existing augmenter data base, as well as any necessary analytical evaluations, scale or full size testing of hardware.

Results of the testing of full scale components will be the topic of discussion in this section of the report. The extent of full scale hardware testing included: evaluation of the Proto 1 existing left hand diffuser, elevon, and endwall blowers in the North American Aircraft Division Columbus plant's Thermodynamics Lab, evaluation of the Proto 1 L/H elevon at a facility at Wright-Patterson Air Force Base, evaluation of the modified improved L/H elevon at WPAFB, and evaluation of improved endwall blowers in North American Aircraft Division, Columbus plant, Thermodynamics Lab.

Additional full scale Coanda development testing was done at WPAFB on a Coanda segment model, and the results of that testing is discussed in Section 6.0 (Coanda Development) of this document. The testing facilities for both the WPAFB test site and North American Aircraft Division, Columbus plant, Thermodynamics Lab are described in Section 10.0 of this report.

9.2 PROTO 1

The evaluation test of the L/H Proto 1 wing components after the Langley hover testing has previously been reported in Reference (33) and will be summarized here.

9.2.1 Diffuser

The L/H wing diffuser was tested in the North American Aircraft Division, Columbus plant, Thermo Lab to obtain the following information:

    a. Nozzle coefficients
    b. Nozzle gaps actual
    c. Nozzle gap expansion
    d. Spanwise nozzle total pressure distributions
    e. Chordwise jet profiles at the nozzle and downstream
    f. Visual indications of flow quality and angularity

The Thermo Lab test set-up is sketched in Figure 9-1. The inboard and outboard endwalls in the vicinity of the diffuser were simulated by contoured aluminum sheets.
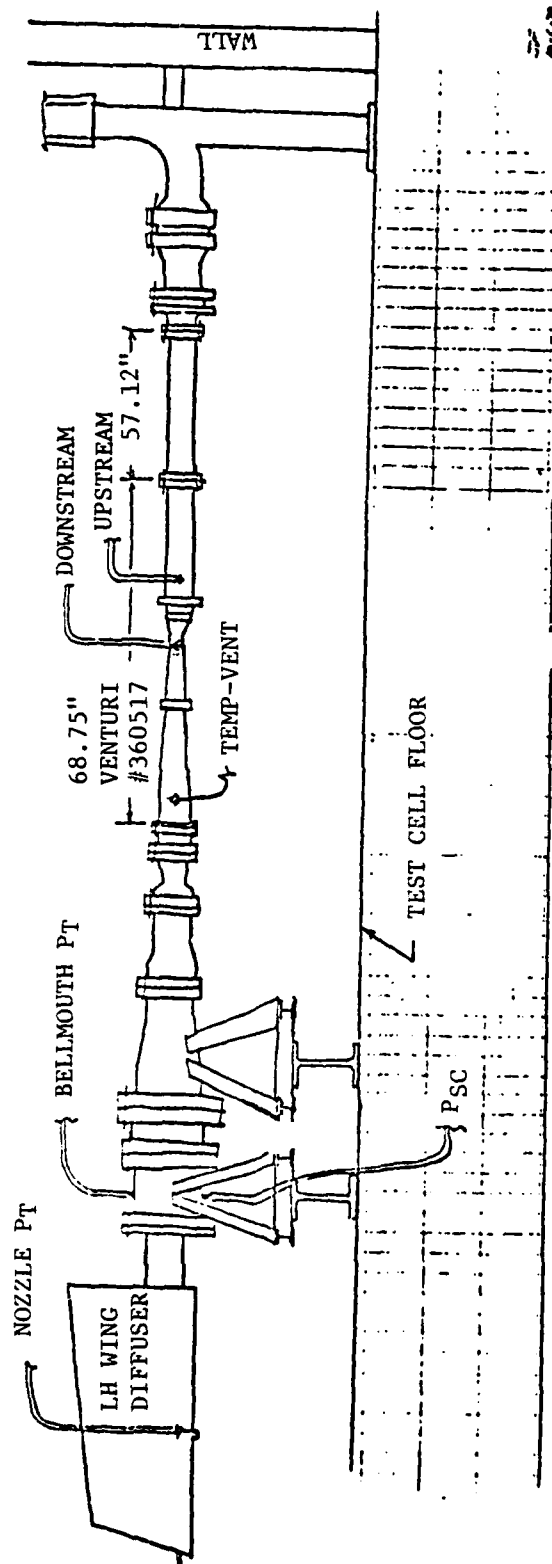
SKETCH OF XFV-12A WING COMPONENT

TEST IN THERMO LAB

Figure 9-1.

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The nozzle total pressure spanwise survey, shown in Figure 9-2, exhibits the same inboard pattern seen in previous tests with the biggest deficiency inboard, immediately followed by tne peak pressure.

The remainder of the survey indicates more uniform spanwise distribution than was indicated in previous tests. The spanwise nozzle pressure distributions were made at nozzle pressure ratios of 1.8, 1.6, 1.5, and 1.2 and were relatively independent of NPR.

Chordwise total pressure profiles were obtained at the nozzle, and at 10t, 20t, and 50t from the nozzle, and also at the trailing edge of the flap. These jet profiles were made at five selected span stations which were located as shown in Figure 9-3. A typical set of profiles is illustrated in Figure 9-4, and a plot of all the measurements is presented in Figure 9-5 in the form of jet peak pressure normalized with nozzle exit pressure (CTE) versus span station. At the three midspan locations, the profiles indicate satisfactory flow to the trailing edge, however, the inboard and outboard pressure profiles reflect the disturbed flow at these locations. The efficiency at 10t appears to be slightly less than had been seen in previous tests. At the outboard span station (107") the efficiencies reflect the hinged triangular panel (see Figures 9-3 and 9-7), which was inclined into the flow for these tests simulating its position on the aircraft.

Flow quality and angularity were investigated using a hand held tuft as a flow visualization aid. Figures 9-6 and 9-7 are photographs of the results. The two inboard surveys show the most outboard spanwise location where steady corner flow was obtained and the most inboard location where steady chordwise flow was obtained. The two outboard surveys likewise indicate the steady flow boundaries. Notes that the disturbed flow region outboard is approximately six inches wide.

Nozzle gap expansion under pressure with cold flow was measured using five linear potentiometers installed at five spanwise locations as shown in Figure 9-8. Results of three runs with the gages installed is presented in Figure 9-9. It can be seen that out to midspan, gap expansion approached four percent at a nozzle pressure ratio of 1.84:1. Outboard the expansion is on the order of one percent. The nozzle gap versus span was also measured without pressure and this data compared to design values is presented in Figure 9-10.

Leakage was investigated with the airflow system blocked at the upstream end of the Venturi, and a flowmeter for low flows installed. The Coanda nozzles were covered with metal plates, sealed, and then covered with tape. The corner nozzles were plugged with tapered wood plugs, sealed, and taped. Leaks on the flap surface appeared to be small. Several leaks in the interface region were identified. A leakage area of 0.46 sq. in. was determined for the diffuser and the flow measurement system. This area was used in determining the nozzle area coefficients presented in Figure 9-11. The leakage area of the wing diffuser itself was approximately 0.24 sq. in.

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DIFFUSER NOZZLE TOTAL PRESSURE SURVEY
NOMINAL NOZZLE PRESSURE RATIO 1.8:1
RUN 8   10-9-78

Figure 9-2.

PRESSURE PROFILE LOCATIONS
PROTO 1 L/H WING DIFFUSER

$Y_{WDF} = 0.0$

X_{WDF}
156.656

107"

NOZZLE C_L

81"

53.5"

25"

4"

X_{WDF}
0.0

10t

20t

50t

TE

PRESSURE PROFILE LOCATIONS

Figure 9-3.

PROTO 1 L/H DIFFUSER JET PROFILES
DOWNSTREAM 161 NOM NPR = 1.8:1

107" SPAN STA

81" SPAN STA

53½" SPAN STA

25" SPAN STA

4" SPAN STA

JET WIDTH - INCHES

P_T - PSIG
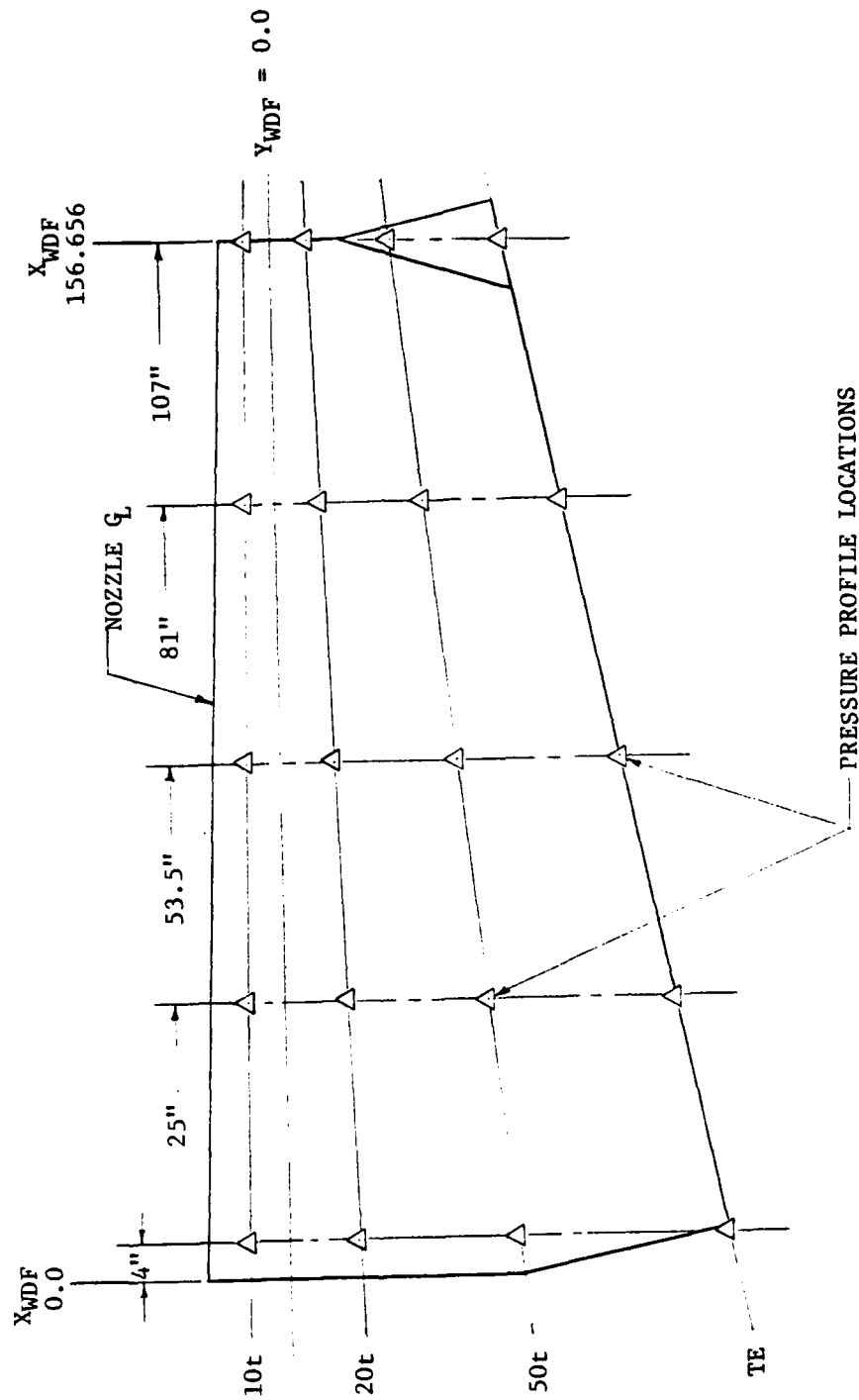
Figure 9-4.

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PROTO 1 L/H WING DIFFUSER
COANDA TURNING EFFICIENCY VS SPAN
NOMINAL NPR = 1.8:1 RUNS 6&7
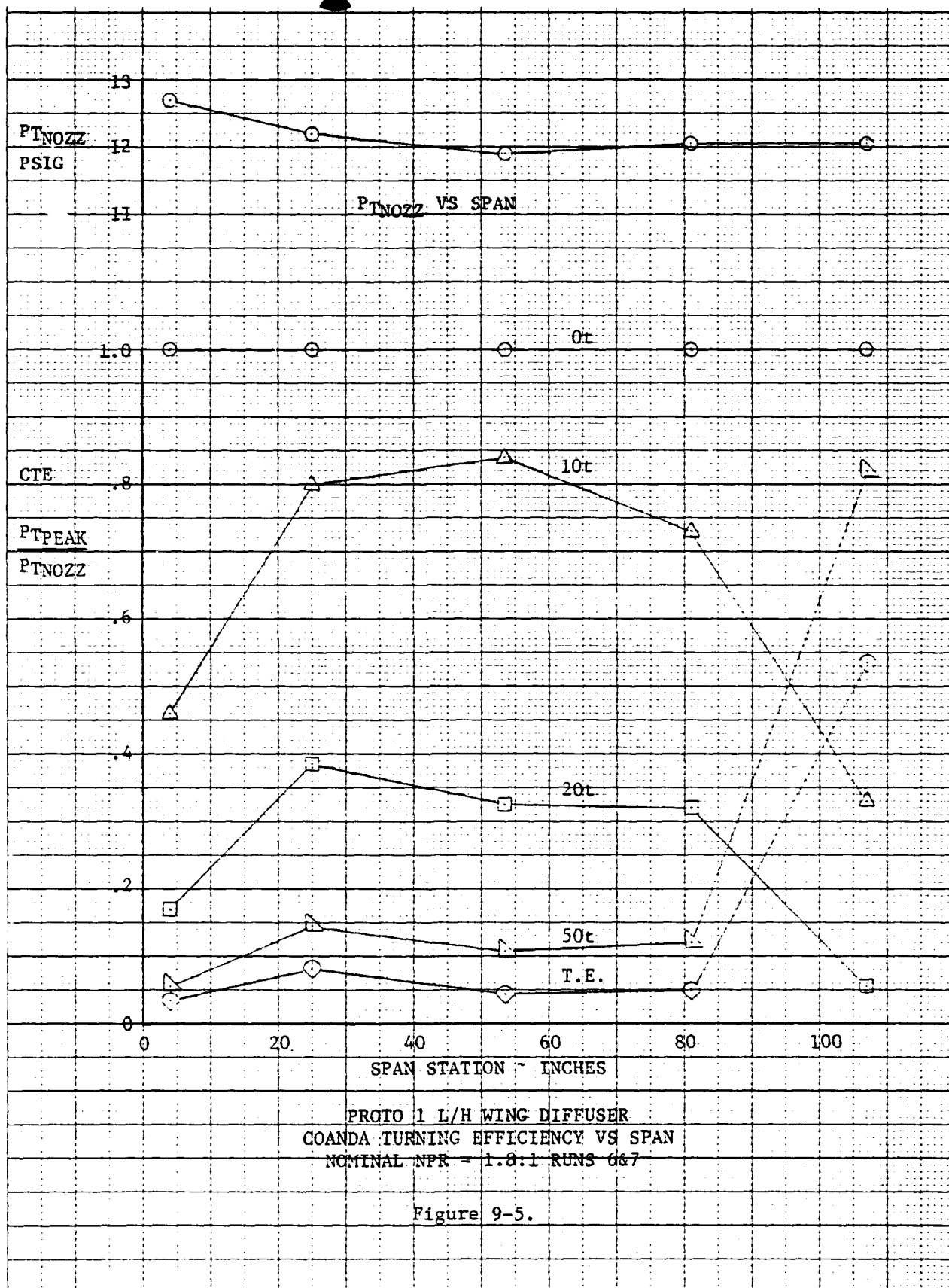
Figure 9-5.

(a) Inboard Corner   (b) Near Inboard Corner
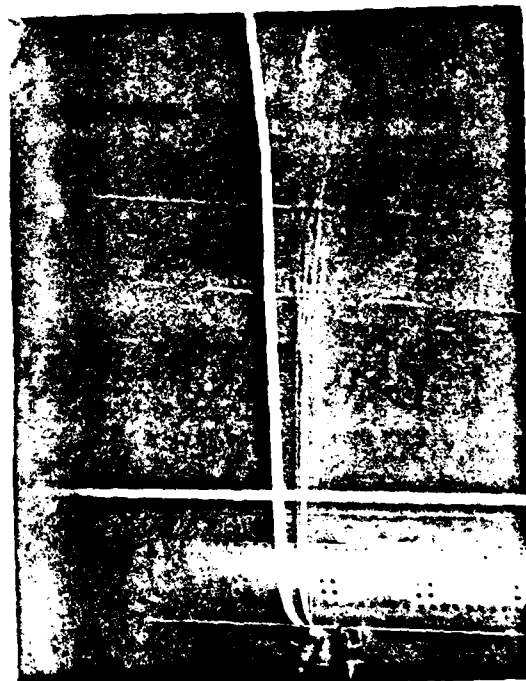
(c) 26" Outboard   (d) 54" Outboard

Figure 9-6.
Flow Visualization
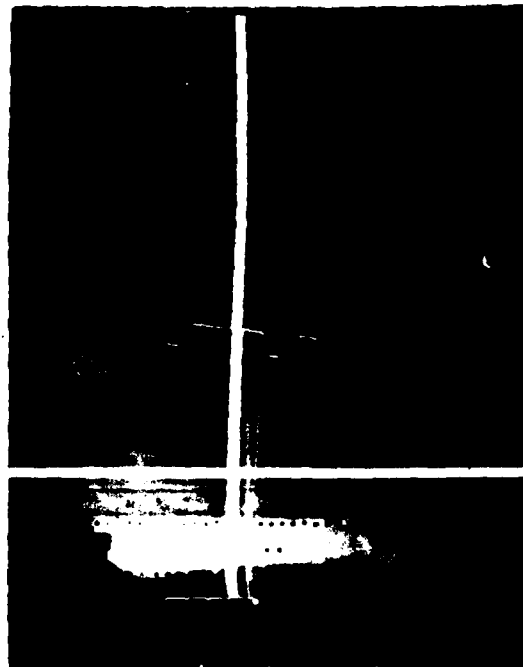
(e) 81" Outboard
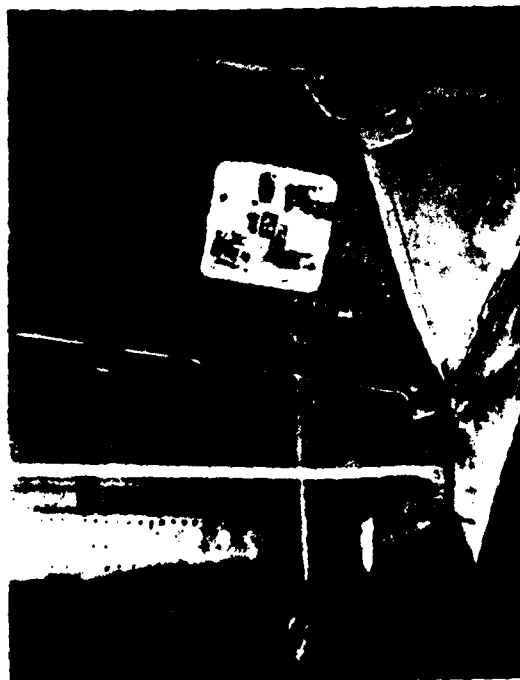
(f) 100" Outboard

(g) Outboard Corner

Figure 9-7.
Flow Visualization
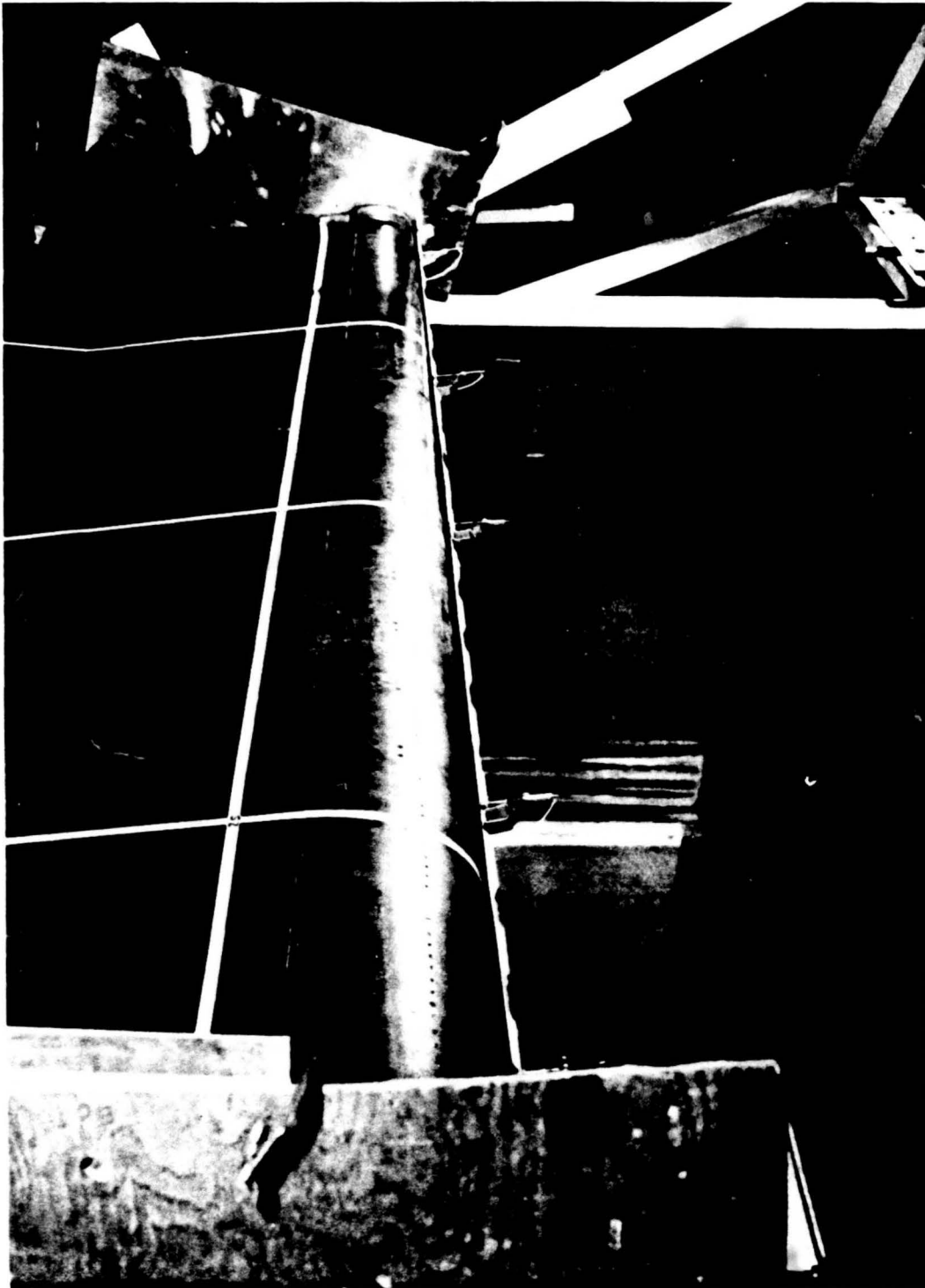
Figure 9-8. Nozzle Expansion "Pots" - Installation on Wing Diffuser

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NOZZLE GAP EXPANSION
PROTO 1 L/H WING DIFFUSER
RUNS 18 19 & 20

GAP EXPANSION - % OF INITIAL GAP

NOZZLE PRESSURE RATIO

○ 15" OUTBOARD
△ 30.5" OUTBOARD
□ 56.5" OUTBOARD
⬡ 77" OUTBOARD
⬦ 101" OUTBOARD

Figure 9-9.

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NOZZLE GAP VERSUS SPAN STATION
PROTO 1 L/H WING DIFFUSER

Figure 9-10.

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C_D = NOZZLE FLOW COEFFICIENT
C_E = NOZZLE PRESSURE EXPANSION FACTOR
(MECHANICAL EXPANSION)

⊙ Thermo Lab isolated surface test data
for period 10-6-78 to 10-13-78.

C_E x C_D ESTIMATED
FROM LAB DATA

LAB GAPPER POTS
ESTIMATED MECHANICAL
NOZZLE EXPANSION

C_E x C_D

XFV-12A
TEST 29
THRU 96

TEST 29 GAPPER GAGE
ESTIMATED MECHANICAL
NOZZLE EXPANSION

NOZZLE COEFFICIENT

1.00

.96

.92

.88

.84

.80
1.2   1.4   1.6   1.8   2.0   2.2   2.4   2.6
NOZZLE PRESSURE RATIO

Figure 9-11.   XFV-12A Left Hand Wing Forward Diffuser Surface
Test Nozzle Area Coefficients

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9.2.2   Elevon

The Proto 1 L/H elevon was also tested in a similar manner as the diffuser utilizing the same test set-up (Figure 9-1). The inboard and outboard endwalls immediately adjacent to the elevon were simulated by flat aluminum sheets.

The nozzle total pressure spanwise survey presented in Figure 9-12 exhibits the same pattern inboard as indicated in previous tests. The lowest nozzle pressure inboard coincides with the outboard corner of the inboard corner blower, then recovers to a level approximately equal to average nozzle pressure in about eight inches. From this point outward, the pressure distribution is relatively uniform as seen in previous elevon tests. As had been found with the diffuser, the spanwise elevon nozzle pressure distribution was relatively insensitive to nozzle pressure ratio. Note that the bolt indicated in Figure 9-12 is a stray bolt which had become lodged in the nozzle, and as it did not interfere with test objectives, no attempt was made to remove it for these tests.

Chordwise total pressure profiles were obtained at the nozzle exit, at 10t, 20t, 50t from the nozzle, and at the trailing edge, at span stations illustrated in Figure 9-13. Figures 9-14 and 9-15 are examples of the jet profiles obtained at the nozzle and at 10t from the nozzle at a nominal nozzle pressure ratio of 1.6:1. At the nozzle the pressure distribution from surface to lip was less uniform than that noted on the diffuser, especially at the inboard corner, however, considering the complex vane geometry required to turn the flow at the inboard corner, the resulting pressure profile was perhaps as good as could be expected with this design. The results of all the jet profile measurements are summarized in Figure 9-16 in the form of turning efficiency (local peak pressure/nozzle exit pressure) versus span. The low values inboard, downstream of the nozzle exit suggest flow angularity in this region, which can also be seen in the flow visualization photographs in Figure 9-17 and 9-18.

Nozzle gap expansion under pressure was measured using the same linear potentiometers used on the diffuser (see Figure 9-8). Results of these measurements are presented in Figure 9-19 in the form of percent gap expansion with pressure relative to the unpressurized reading for several span stations. Unlike the diffuser, the gap expansion is relatively linear with nozzle pressure ratio and uniform along the span of the nozzle, reaching a value of approximately four percent expansion at the highest nozzle pressure ratio tested (1.82:1). The actual nozzle gaps measured with no pressure versus span are presented in Figure 9-20 and the results compared to the design values.

Leakage was investigated with the airflow system blocked at the upstream end of the six-inch Venturi, and air was supplied through a two-inch Venturi and in turn through the endwall blowing feed pipe. The two inch Venturi was used, instead of the low-flow flowmeter used in the leakage test of the diffuser, in order to obtain a wider range of test pressure ratios. The nozzles were sealed in the same manner as the diffuser was, in that the Coanda nozzles were covered with metal plates, sealed, and

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ELEVON NOZZLE TOTAL PRESSURE SURVEY
NOMINAL NOZZLE PRESSURE RATIO 1.7:1
RUN-11 · 10-27-78

Figure 9-12.

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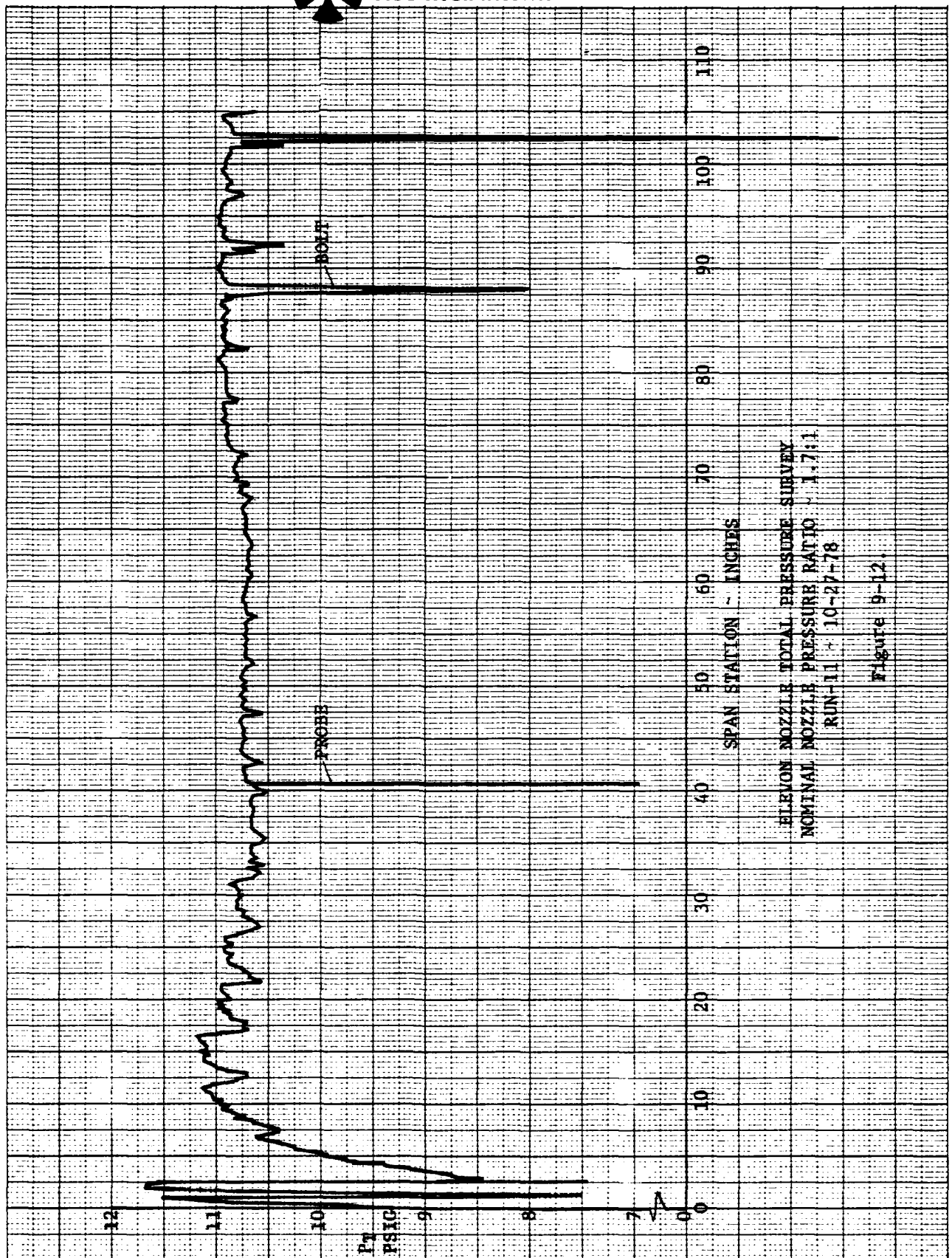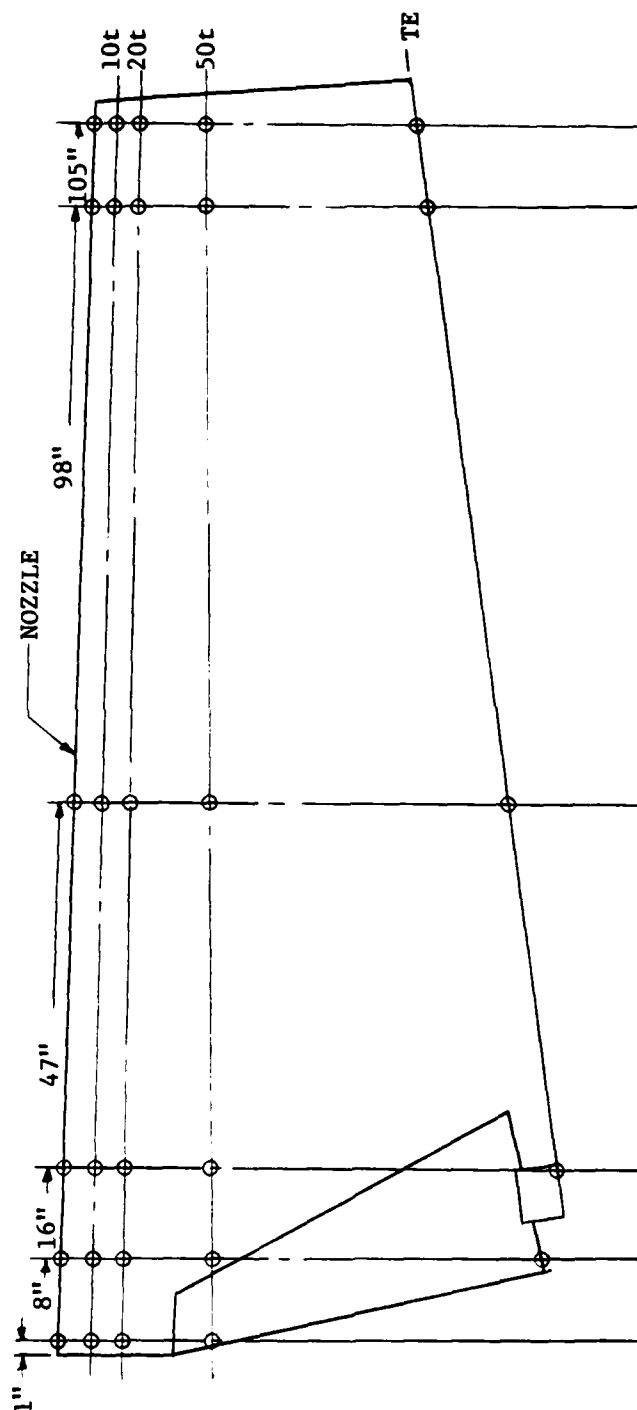
Figure 9-13. Pressure Profile Locations – Proto 1 L/H Wing Elevon
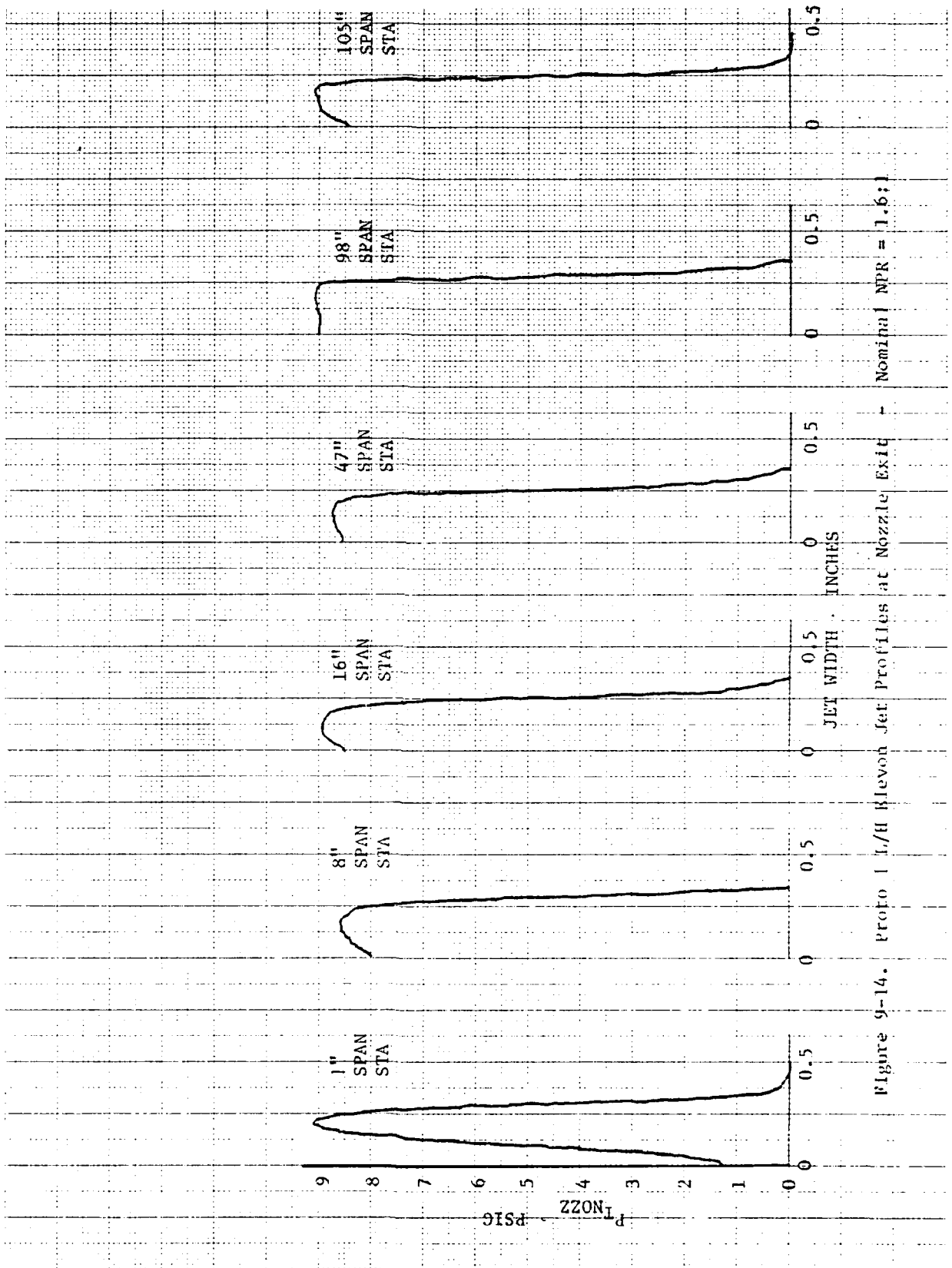
Figure 9-14. Proto 1 L/H Elevon Jet Profiles at Nozzle Exit - Nominal NPR = 1.6:1
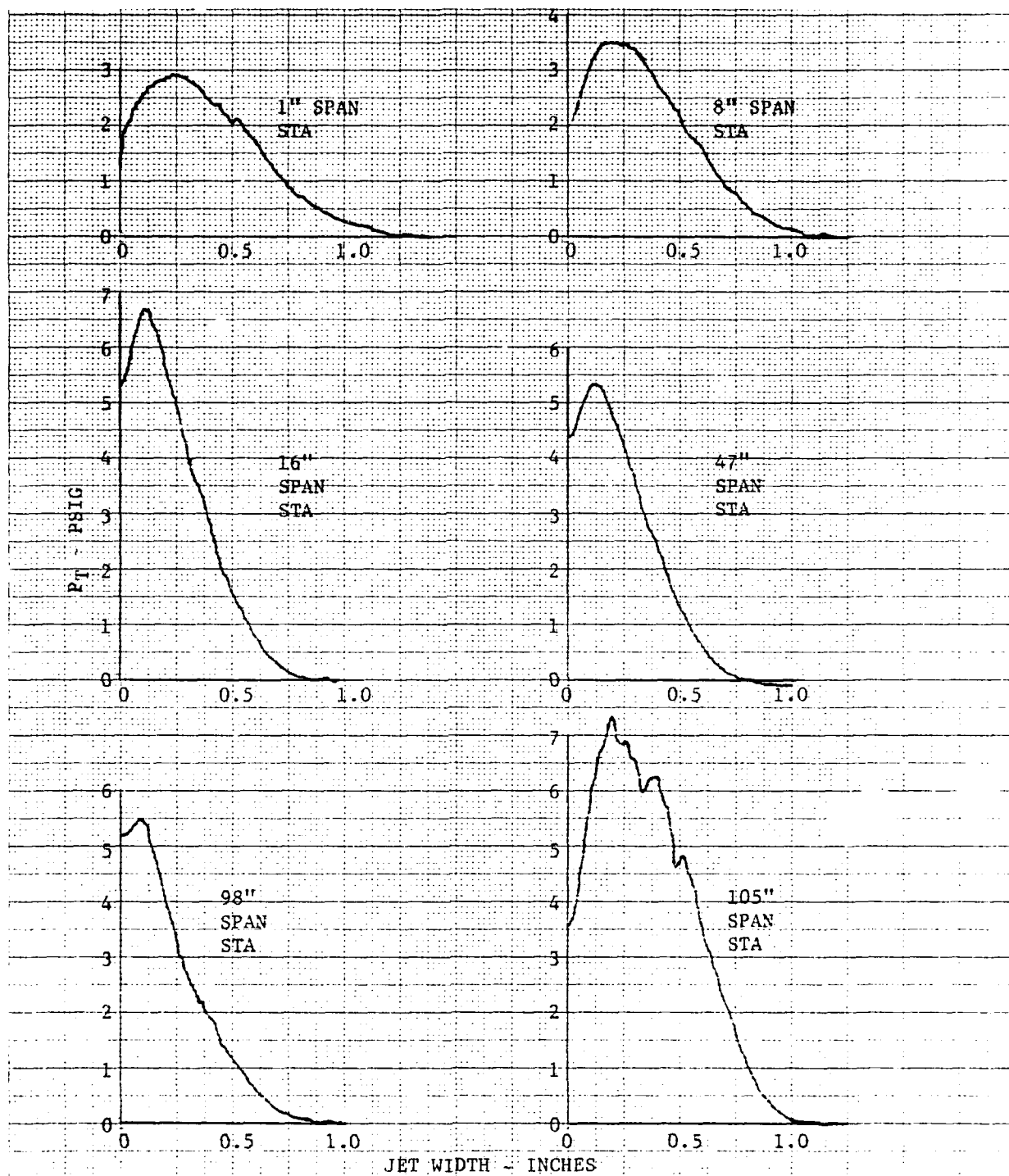
Figure 9-15. Proto 1 L/H Wing Elevon Jet Profiles Downstream 10t Nominal NPR 1.6:1
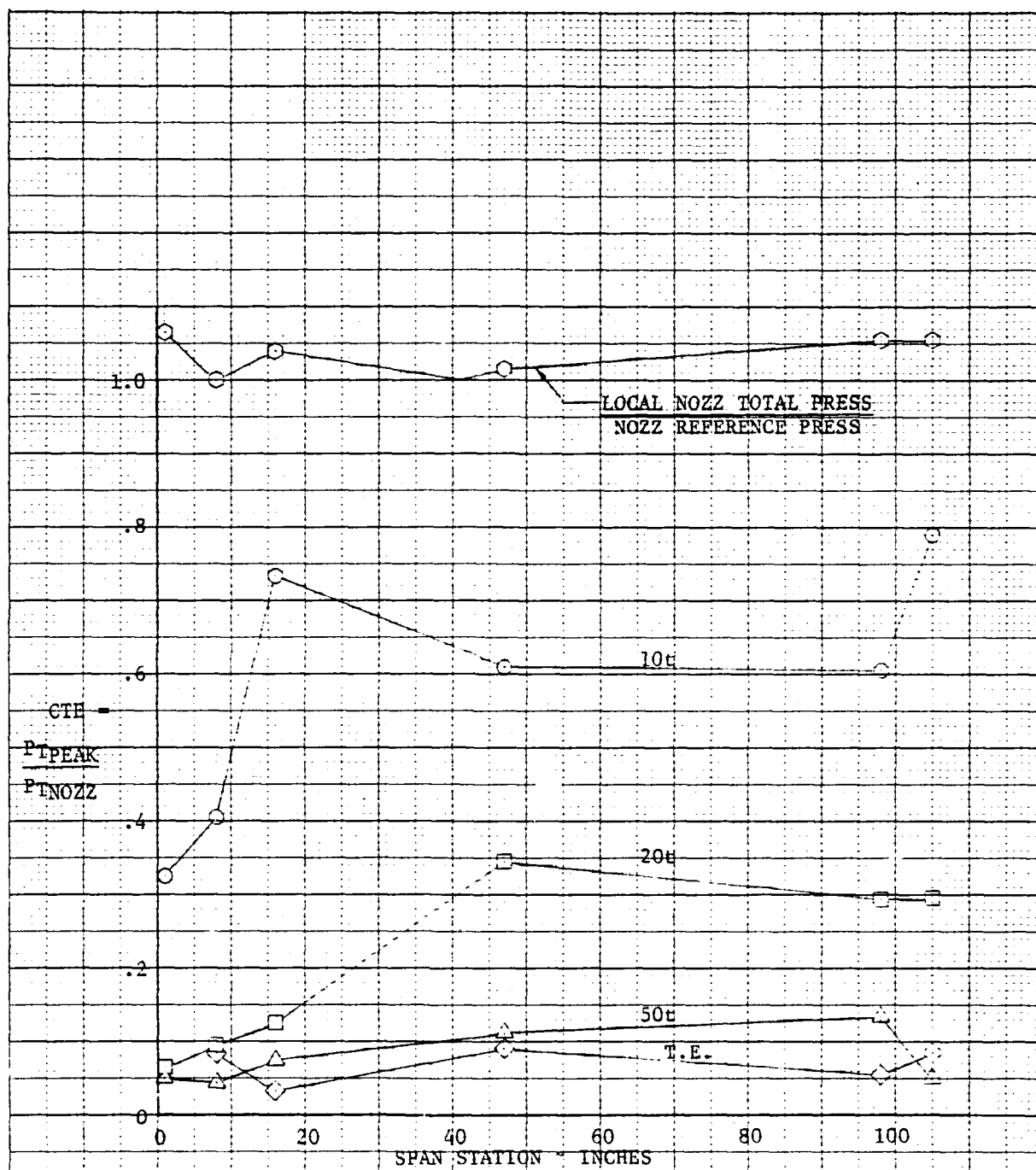
Figure 9-16. Coanda Turning Efficiency Vs Span, Proto 1 L/H Wing Elevon, Nominal Pressure Ratio = 1.6:1

Inboard Corner

6" Outboard

8" Outboard

14" Outboard

Figure 9-17.
Flow Visualization

47" Outboard

98" Outboard

102" Outboard

Outboard Corner

Figure 9-18.
Flow Visualization

9-21

NOZZLE GAP EXPANSION
PROTO 1 L/H WING ELEVON
RUNS 7, 8, & 9

○ 7" OUTBOARD
△ 28.6" OUTBOARD
□ 51" OUTBOARD
◇ 74" OUTBOARD
⊕ 99.2" OUTBOARD

Figure 9-19

Rockwell International

Figure 9-20. Nozzle Gap Versus Span Station, Proto 1 Left Hand Wing Elevon
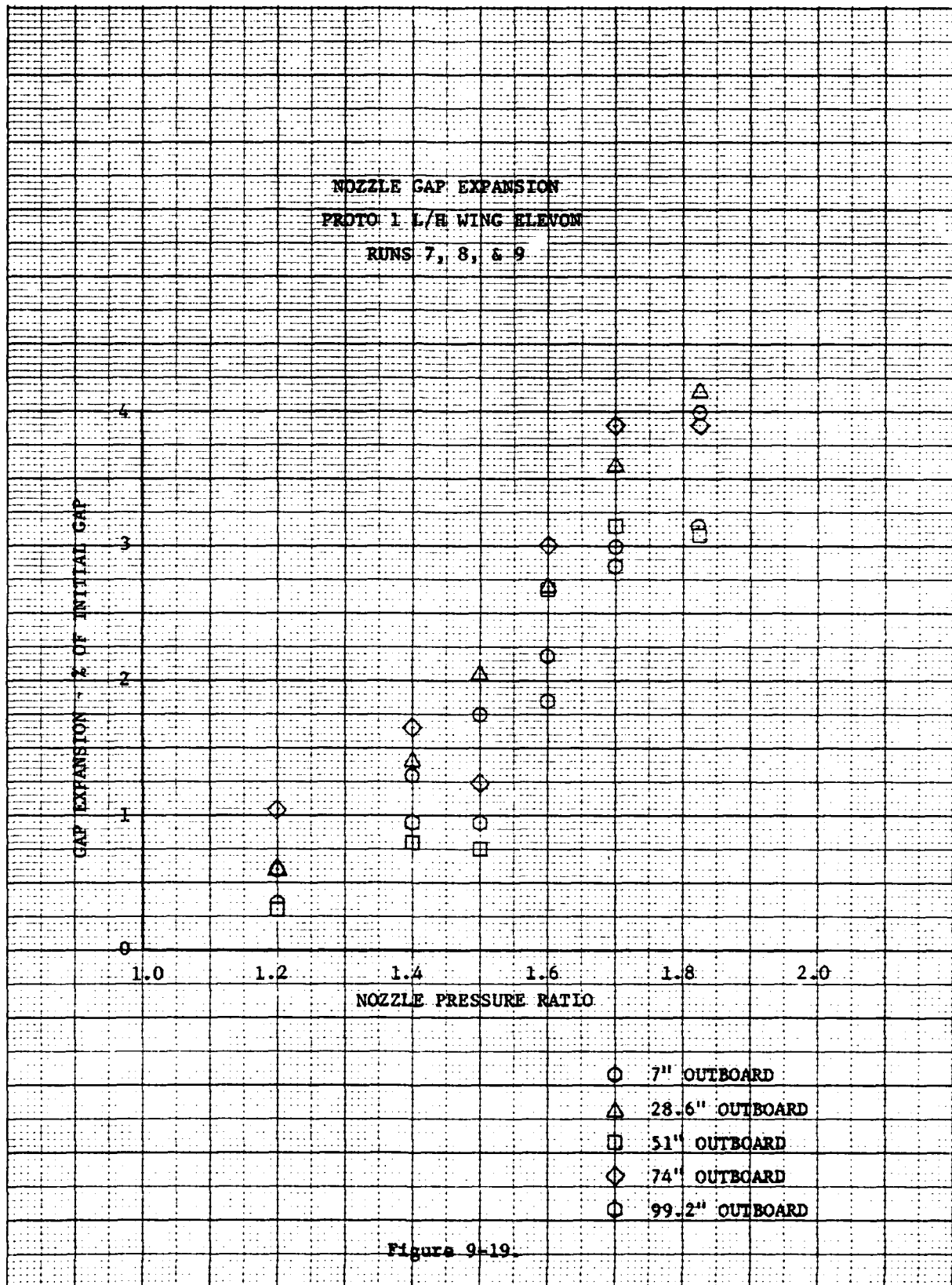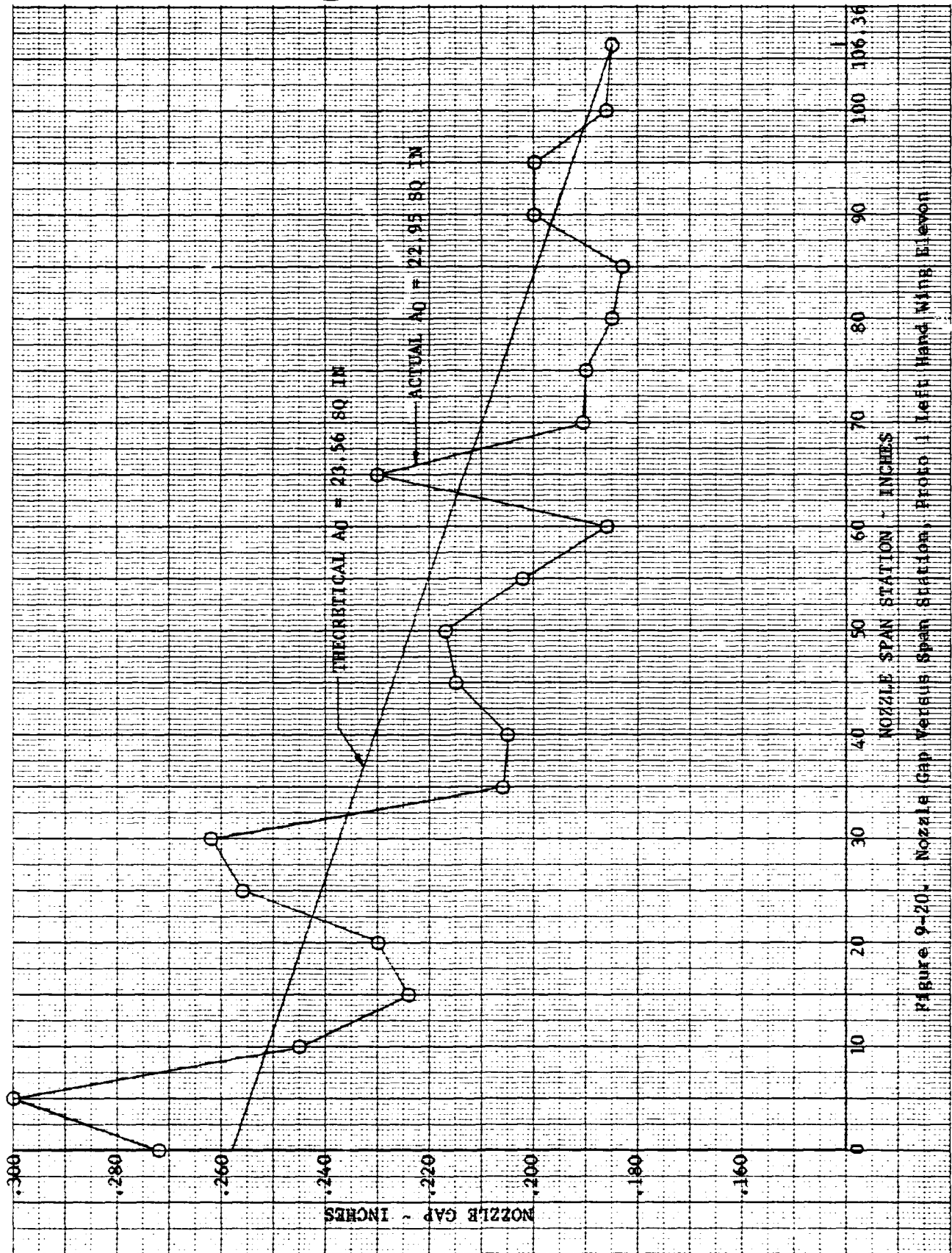
covered with tape. Leaks on the flap surface appeared to be minor. Leakage from under the upstream edge of the inboard corner blower hood was quite apparent. With the nozzle unplugged, such leakage would tend to be less, however, any leakage flow in this region tends to act against the Coanda nozzle flow. Such flow, and the edge of the hood itself, are adverse to improving corner flow. Leakage at the inboard and outboard edges of the elevon was quite apparent.

From the measurements an equivalent leakage area of 0.75 sq. in. was determined for the elevon and the flow measurement system. This area was used in determining the nozzle area coefficients presented in Figure 9-21. The leakage 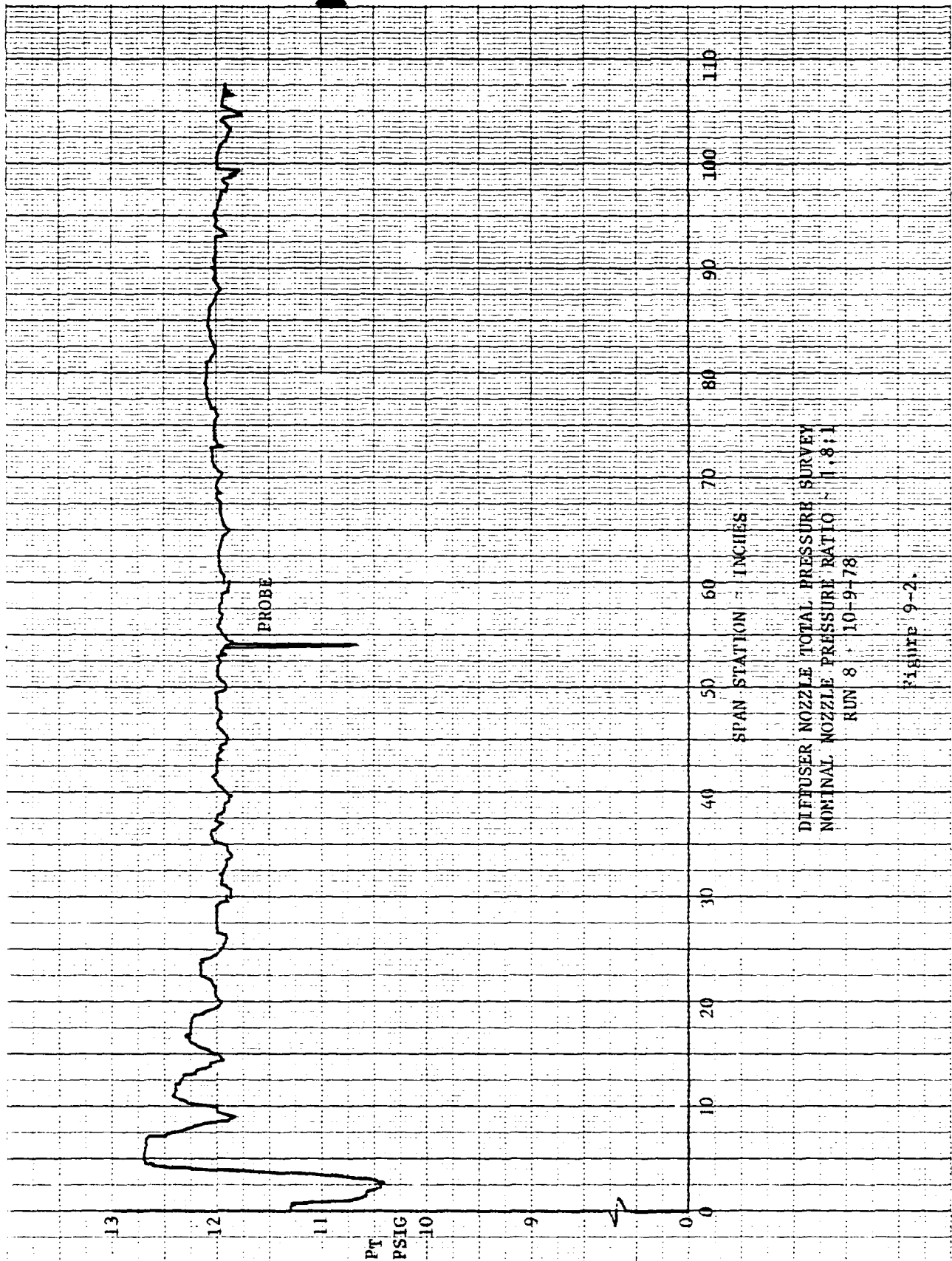area of the wing elevon itself is approximately 0.53 sq. in., twice that of the wing diffuser. The overall nozzle coefficient presented in Figure 9-21, which takes into account the combined effect of the discharge characteristics of the nozzle, leakage, and nozzle area change, is within two percent of previous lab test results. With the nozzle expansion contribution isolated, the nozzle flow coefficient is seen to be less than unity as expected at pressure ratios of interest. The typical flat characteristic of the overall coefficient with pressure ratio likewise echoes previous lab test results.

9.2.3   Isolated Wing Endwall Blowing Nozzle Tests

The inboard and outboard wing endwall blowing nozzles which were part of the aircraft configuration tested at NASA Langley were tested as isolated components in the Thermodynamics Laboratory in order to verify the nozzle coefficients used in the Langley performance computations and to determine areas of potential improvement. As a consequence of these initial tests, design modifications were tested. Results of these additional tests indicated that improved flow characteristics were obtained with the modified inboard and outboard nozzles. Recommendations for further improvement had been made at the time testing was suspended. Further testing was deferred when the ongoing aircraft design studies being conducted in parallel indicated that the simpler augmenter concept developed would in turn enable simpler, more efficient endwall blowing nozzles to be incorporated in the revised augmenter design. The lessons learned in these tests, however, are generic to the design of any endwall blowing system.

9.2.3.1 Outboard Endwall Blowing Tests - Initial tests were conducted with the basic design, a straight Coanda slot nozzle, and with a hood added to the basic nozzle. The hood was added in the course of the tests at NASA Langley to direct more flow aft. The results of these lab tests, both with and without the hood, indicated that flow coefficients were similar to those used at Langley, and while flow directly downstream of the nozzle was satisfactory, flow in the fore and aft corners, formed by the diffuser flap and elevon, respectively, was unsatisfactory. Based on both pressure surveys and flow visualization, flow in the elevon corner was essentially non-existent.

To improve flow in the corners, the straight nozzle was replaced by one in which the ends of the straight nozzle were canted to point to the forward and aft corners. Nozzle geometry is presented in Figure 9-22.

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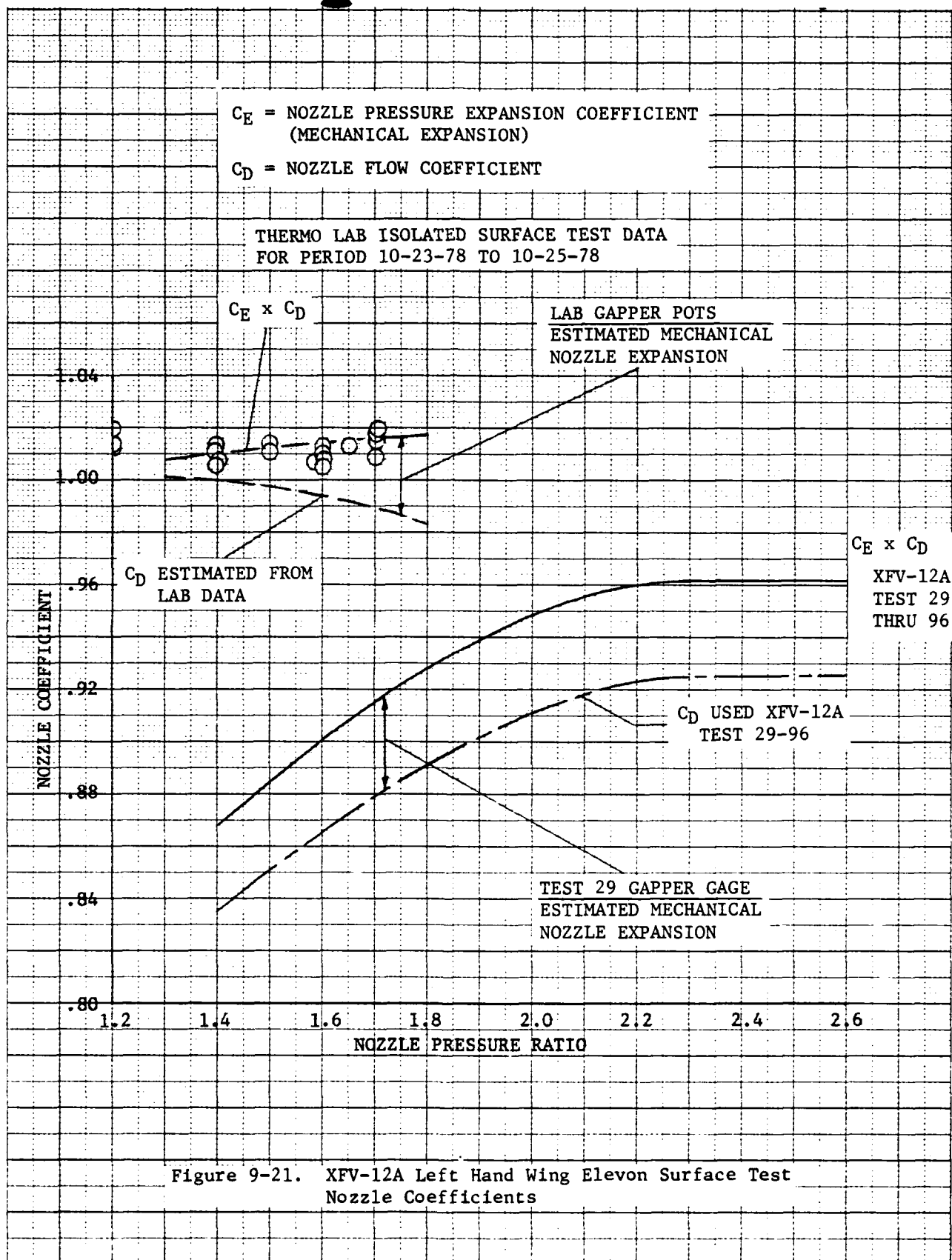
Figure 9-21. XFV-12A Left Hand Wing Elevon Surface Test
Nozzle Coefficients

	PRIMARY	AUXILIARY SLOT	EXTENDED ELEVON
TOTAL LENGTH (IN.)	9.81	0.75	1.31
AVG GAP (IN.)	0.205	0.255	0.155
⋇ TO $\mathcal{C}_L$ ⊥	0°	52°	28°
AREA (IN.²)	2.029	0.191	0.203
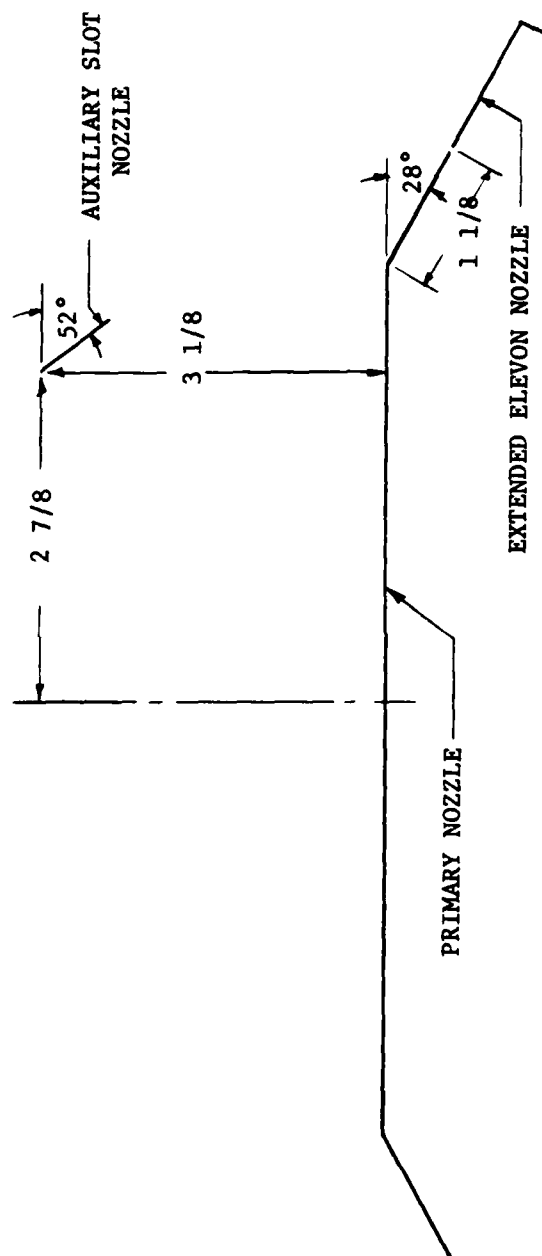
Figure 9-22. Outboard Endwall Nozzle Dimensions

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Results of initial tests of the modified nozzle indicated that flow in the diffuser corner was satisfactory, but that flow in the elevon corner was not. Addition of an auxiliary slot nozzle turned more toward the elevon, located 3-1/8 inches below the primary nozzle (to minimize interference with nozzle structure) provided a small, insufficient improvement in the corner flow. With the addition of an auxiliary nozzle which effectively extended the primary nozzle into the elevon corner, satisfactory flow in the endwall-elevon corner was obtained. The simulated extended elevon corner nozzle and auxiliary slot nozzle are shown in relation to the primary nozzle in Figures 9-22 and 9-23. Pressure profiles obtained with this configuration are compared with the results of the tests of the Langley test nozzle configuration in Figure 9-24. These show the significant increase in pressure obtained in the elevon corner and the more symmetrical fore-to-aft pattern. At the conclusion of this test series, it was recommended that the primary nozzle be extended to the elevon corner and be subjected to performance verification tests with and without the auxiliary slot nozzle.

The primary design guideline resulting from these tests is the necessity for extension of the endwall nozzle to the flap/endwall corners, for in order to minimize separation, the total endwall surface must be subjected to endwall blowing.

9.2.3.2 <u>Inboard Endwall Blowing Tests</u>. The inboard endwall nozzle was initially tested with and without the hood that had been added at Langley to improve flow in the elevon-endwall corner. Test results indicated that the flow coefficient was similar to that used in the Langley tests. They also indicated that flow in the elevon corner with the hood installed was satisfactory at the lower end of the elevon but that a region of no flow existed immediately downstream of the augmenter throat. With the hood removed, flow in the corner was wholly unsatisfactory, as indicated by the randomly directed tufts seen in Figure 9-25. As seen also in Figure 9-25, however, with the hood installed, a separated flow region existed on the endwall downstream of the nozzle-hood intersection. Flow in the diffuser-inboard corner was similar to the elevon corner flow in that it was satisfactory at the lower end of the diffuser flap but inadequate near the throat.

In order to obtain satisfactory flow characteristics, separate nozzles, designed to direct flow into each corner, were installed in the endwall in addition to the basic nozzle. These are shown in Figure 9-26. Note in the test set-up that 2 x 4's were used to simulate the diffuser and elevon flap intersections with the endwall for this series of tests. A traversing rake was used to obtain the pressure distribution, at a fixed distance downstream of the primary nozzle exit, between the elevon and diffuser flaps. Typical of the distributions obtained are those presented in Figure 9-27, which were obtained at a distance 40, 70, and 100 throat heights downstream, 1/4 inch and 1/2 inch above the endwall surface. These distributions indicate that flow in the diffuser flap corner is unsatisfactory in the upstream region, that pressures are low on the elevon side of the endwall and that peak pressures are obtained downstream of the primary nozzle.

EXTENDED ELEVON CORNER NOZZLE

AUXILIARY SLOT NOZZLE

Figure 9-23.   Simulated Extended Elevon Corner Nozzle & Auxiliary
Slot Blower

THERMO LAB ISOLATED TEST
NOMINAL NOZZLE PR = 1.9

PROFILES AT 100t
(APPROXIMATELY 21 IN. DOWNSTREAM OF NOZZLE)

— — — LANGLEY CONFIGURATION
———— REVISED CONFIGURATION
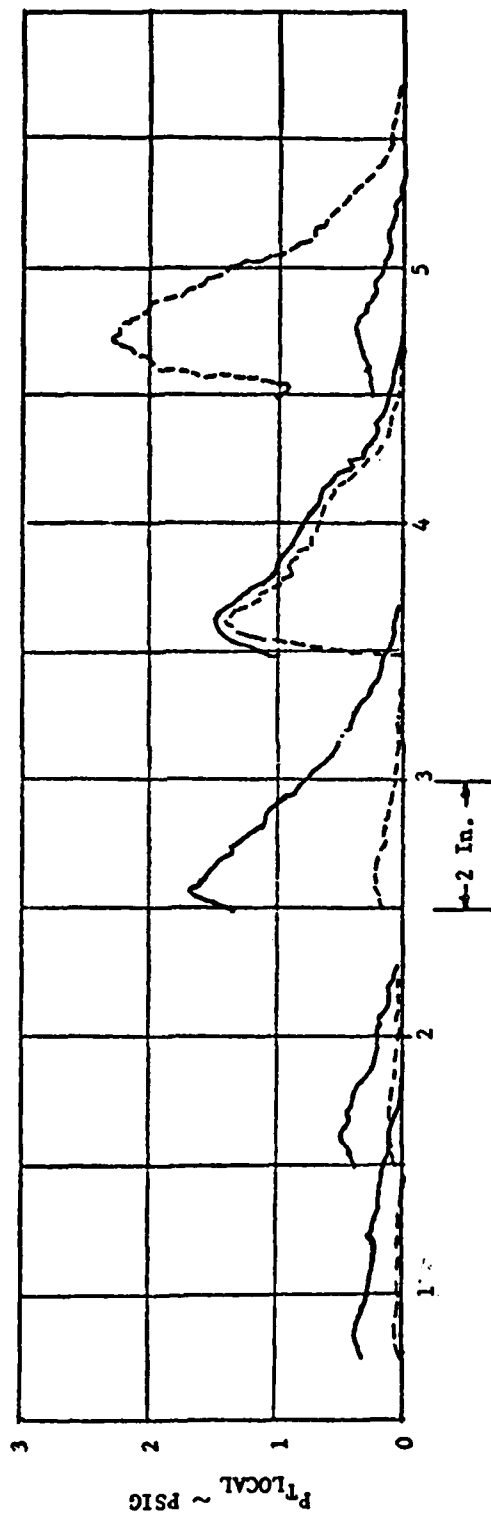
Figure 9-24.
PRESSURE PROFILES
OUTBOARD ENDWALL BLOWING
XFV-12A WING
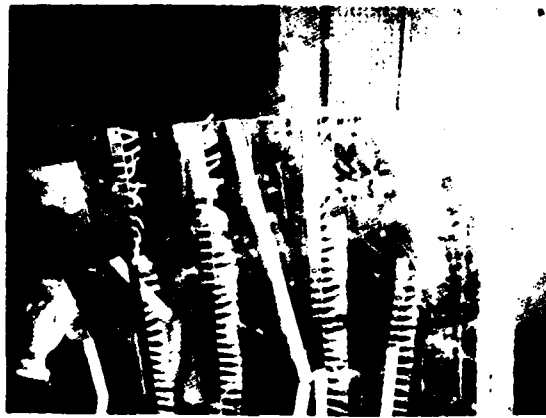
AFT CORNER WITH HOOD REMOVED

AT FORWARD CORNER

NEAR FORWARD CORNER

Figure 9-25. Inboard Endwall Flow Visualization
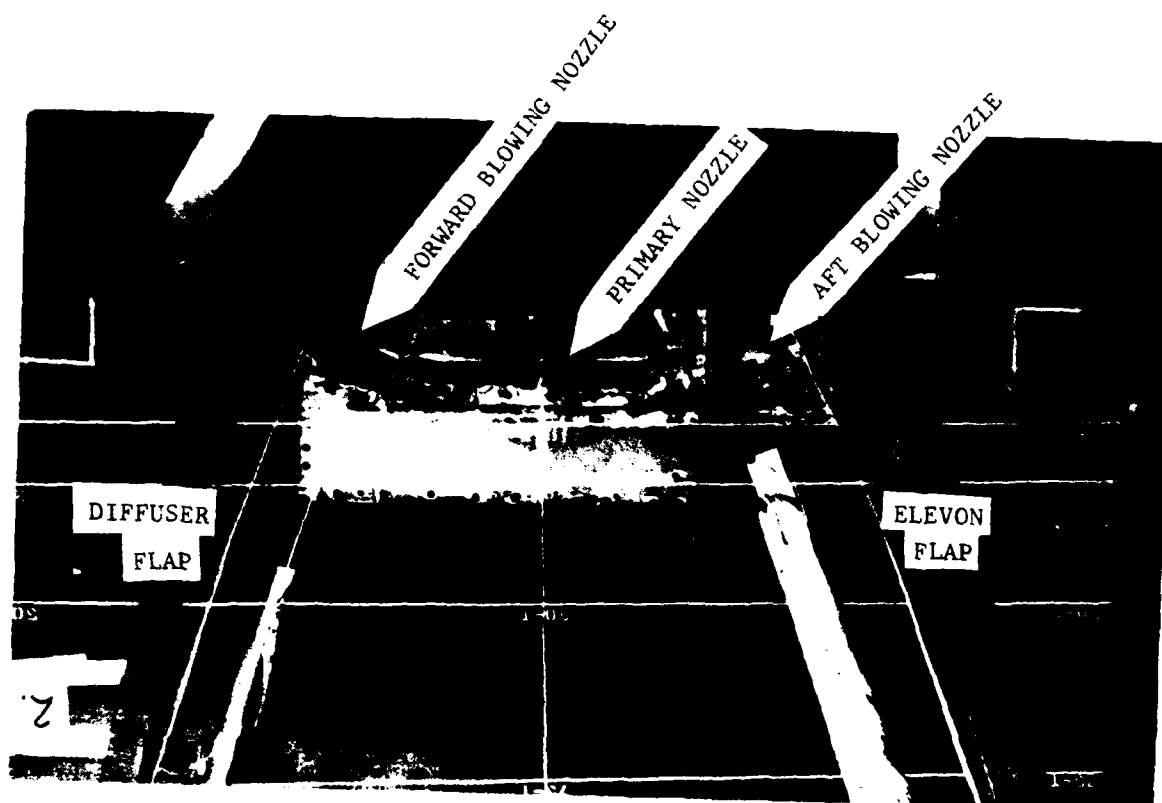
Figure 9-26. Nozzle System Close-up

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FIGURE 9-27.

TRAVERSING RAKE SURVEYS

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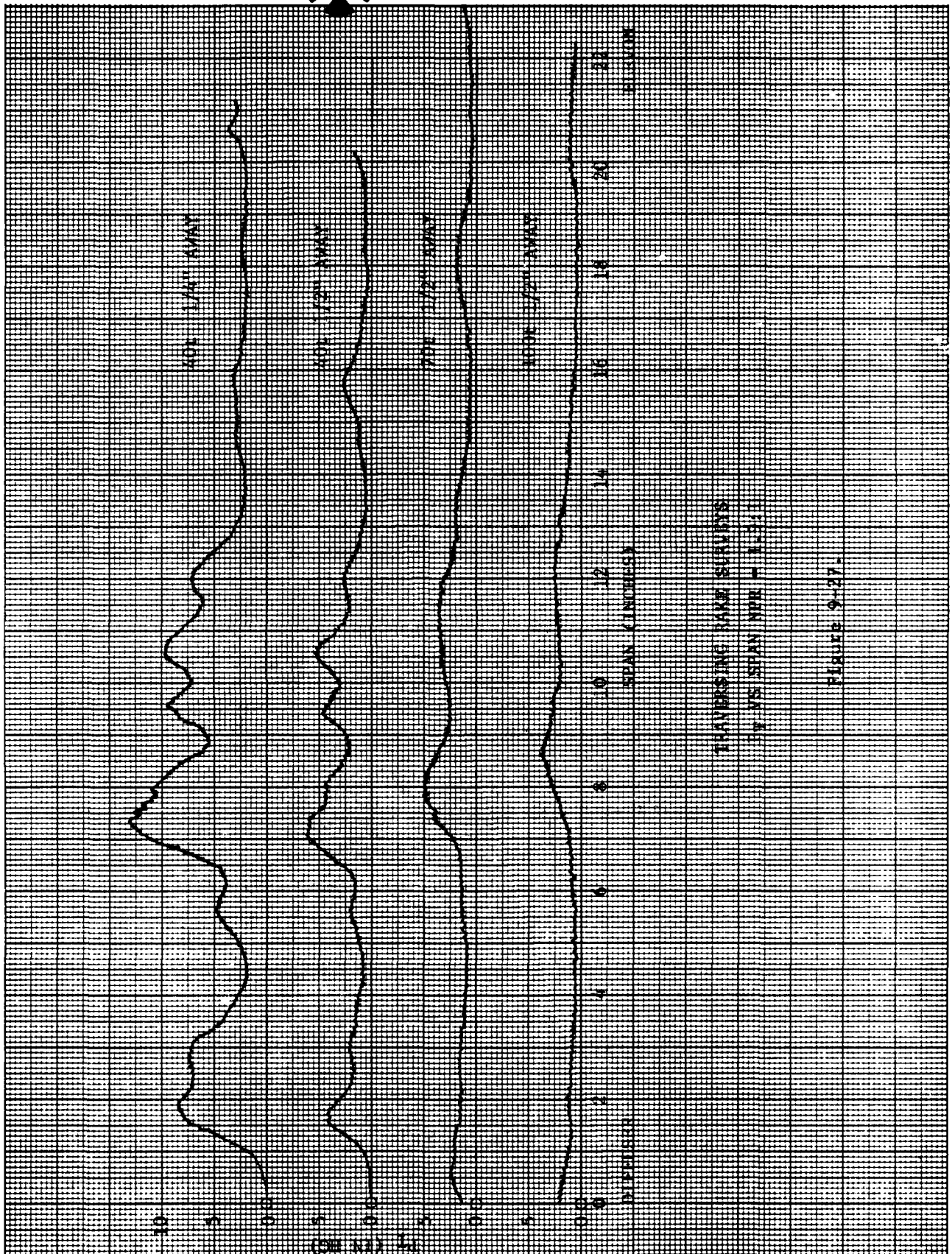
In order to improve flow in the diffuser corner, it was recommended that the nozzle exit angle be increased. To improve the pressure level and distribution on the aft portion of the endwall, it was recommended that the separate aft-blowing nozzle be eliminated and that the primary nozzle be extended aft to the elevon-endwall corner. These recommendations are consistent with those for the outboard endwall.

9.3   IMPROVED ELEVON

Results of the Coanda segment model test which are documented in Reference (20) and summarized in Section 6 of this document indicated a number of potential improvements which could be made to the elevon, namely: (1) change the external Coanda shape to a log spiral with a K factor of 1.3, (2) relocate the inlet feed duct relative to the nozzle exit plane, (3) incorporate primary and dedicated nozzle vanes.

Prior to incorporating the improvements, developed during the Coanda segment model test, on the existing XFV-12A elevon, it was considered prudent to develop a baseline data base on the existing elevon at the WPAFB facility, to provide a good before and after comparison of the elevon modifications. The results of this test are documented in Reference (34) and essentially confirmed the isolated performance tests conducted at Columbus Aircraft Division's Thermo Lab, (Reference 33).

After modification of the existing wing elevon with the improvements developed during the Coanda segment model test the elevon was returned to WPAFB for evaluation of the modifications. Results of this test, which are documented in Reference (35), showed the modifications incorporated resulted in the following improvements: (1) Airflow quality at the inboard corner was significantly improved, (2) turning efficiency was improved along the entire span, (3) elimination of irregularities in the spanwise total pressure distribution, (4) expansion of the primary nozzle gap was markedly reduced.

9.3.1   Test Article

A comparison of the design parameters between the Proto 1 elevon and the modified elevon is tabulated in Table 9-1. All design changes to the elevon assembly are documented in the drawing of Reference (36) and a comparison of the Coanda shapes and feed duct location may be seen in Figure 9-28. Details of the internal nozzle vanes and an overall view of the modified elevon in the WPAFB test facility are shown in Figures 9-29 and 9-30, respectively. A listing of significant design changes incorporated are listed below:

a. The original Coanda shape, which nearly represented a K=1.0 spiral was replaced with a K=1.3 log spiral.

b. Three cascade vanes and a bullet fairing were added to the entrance of the inboard dedicated nozzle.

c. The nozzle lip convergence angle was increased from approximately 0° to a nominal 7.5°.

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Table 9-1 Design Parameters

	INBOARD	EXISTING ELEVON	OUTBOARD	INBOARD	IMPROVED ELEVON	OUTBOARD
A_0		23.543			23.474	
R_0/t	8.319		4.922	11.283		9.630
R_i/t	3.991		3.515	4.00		4.00
S/t Throat	27.151		16.835	34.437		29.387
S/t Exit	169.700		146.000	158.106		187.544
Feed Duct Area		50.266			50.266	
Feed Duct M_n		.25			.25	

TRANSITION DUCT

Y_{MGL} 0

Z_{MGL} 0

X_{MGL} 35.56

PROTO NO. 1

K = 1.30 SPIRAL
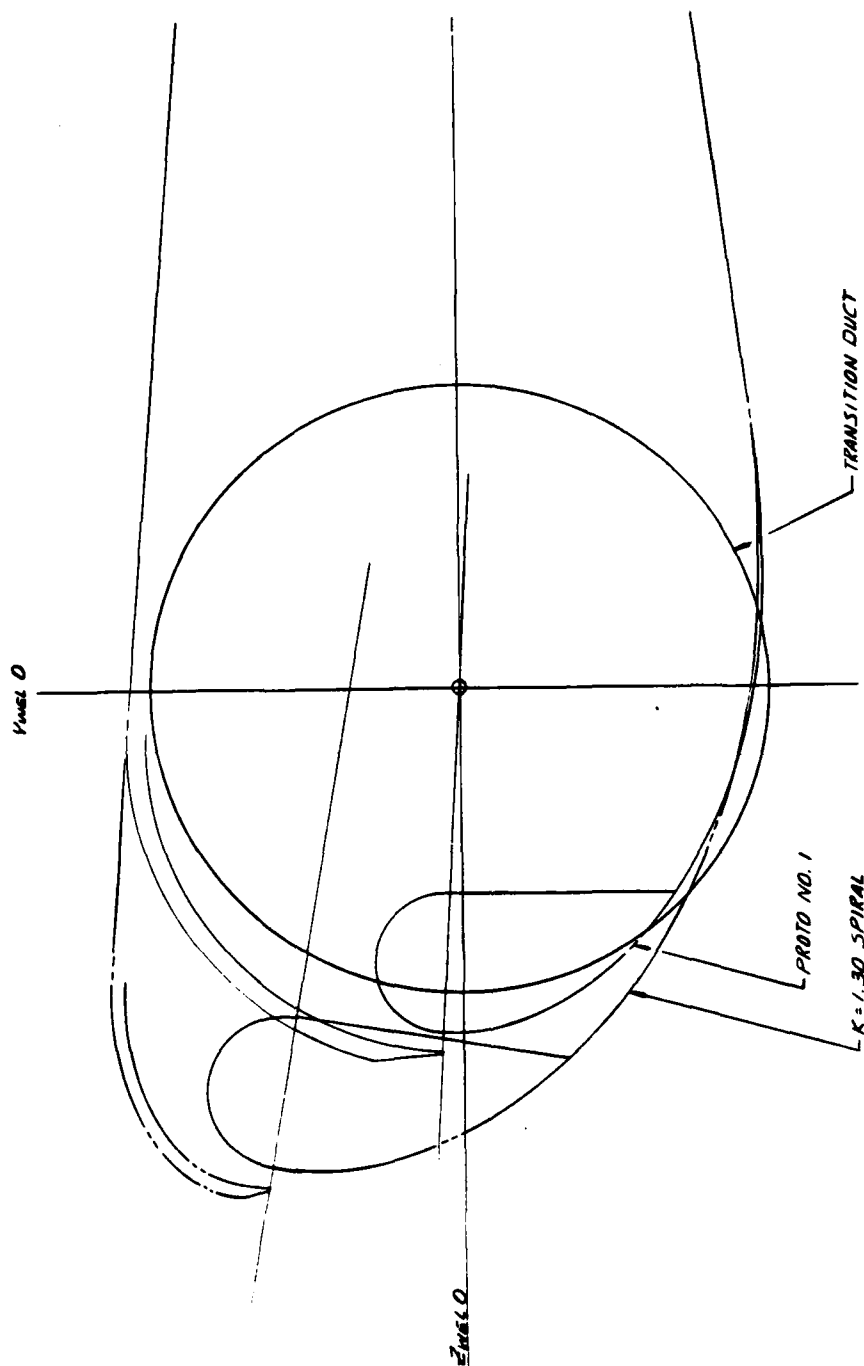
Figure 9-28. Comparison of Coanda Shapes
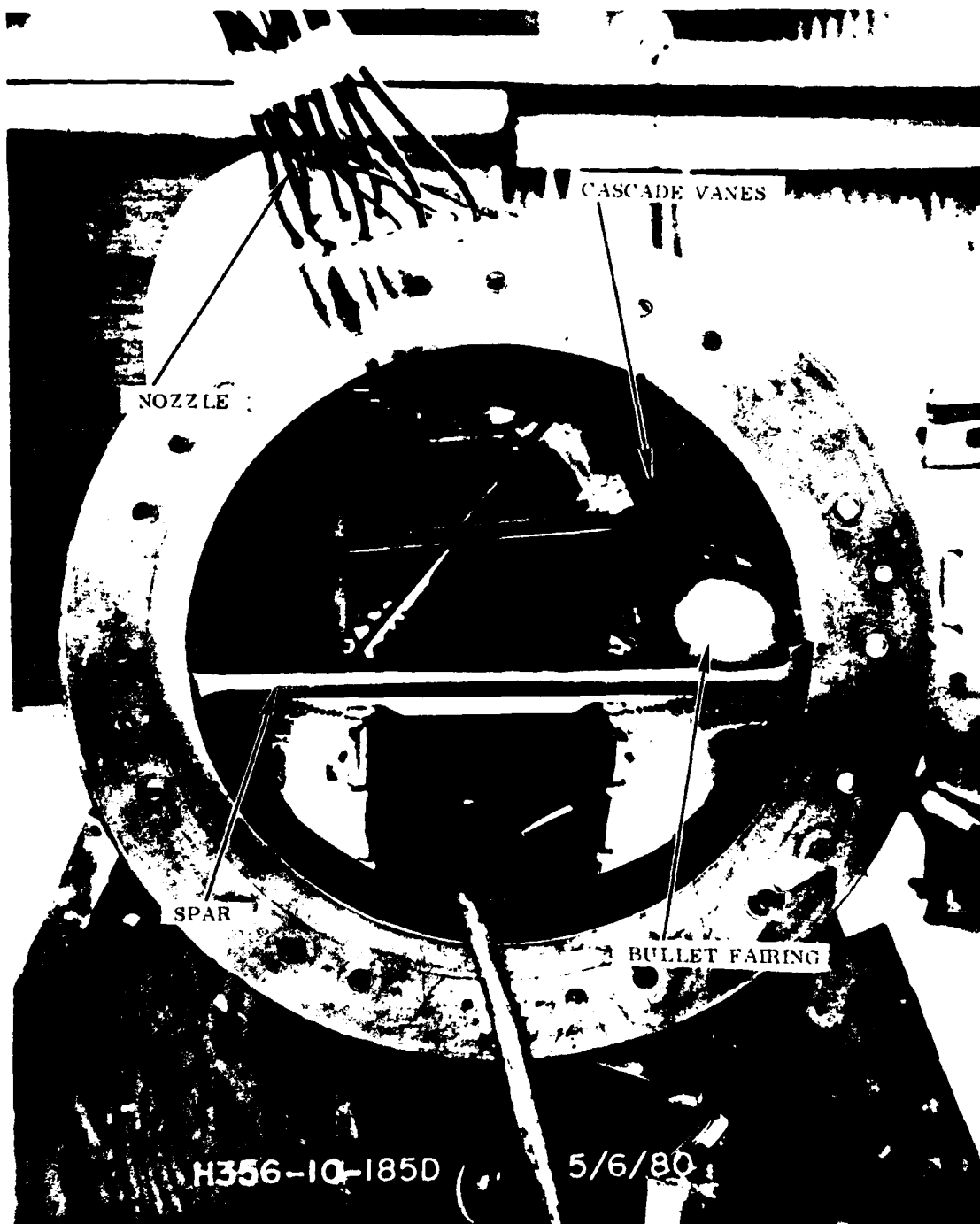
Figure 9-29. Elevon Feed

Figure 9-30. Elevon Installation in Test Stand

FORM 351-F   REV 5-77

d. The primary nozzle gap taper ratio (GAP O.B./GAP I.B) was changed to .5 to agree with the throat taper ratio.

e. The nozzle internal convergence at 1t was increased to 16°.

9.3.2 Test Results

The improved modified elevon was evaluated by means of the following measurements and criteria:

1. Nozzle Cap Measurements
2. Primary and Dedicated Nozzle Spanwise Pressure Surveys
3. Flow and Velocity Coefficients
4. Turning Efficiency
5. Flow Visualization for Flow Angularity.

Results of nozzle gap measurements of the Proto 1 elevon and the modified elevon may be compared in Figures 9-31 and 9-32. Both sets of measurements were made after pressurization, and show considerably better gap vs. span uniformity for the modified elevon. The improved internal nozzle construction was also evident in the nozzle gap expansion vs. nozzle pressure ratio data presented in Figure 9-33. Similar improvements were noted in the dedicated nozzles..

Figures 9-34 thru 9-36 are plots of nozzle total pressure vs. span comparing results of the Proto #1 vs. improved elevon. Considerable improvement in pressure irregularities were realized, especially at the inboard and outboard ends. Similar improvements, in the pressure distributions of the dedicated nozzles, were obtained, especially on the outboard dedicated nozzle, as can be seen in Figure 9-37.

Figures 9-38 and 9-39 present the results of testing the elevons for flow (C_D) and velocity (C_V) coefficients. Although the plots show differences in character vs. NPR, little or no improvement was considered in these two parameters.

The Turning Efficiency (CTE) as defined, CTE = $P_{Tpeak}/P_{Tnozzle}$, is a measure of the pressure remaining in the jet after traveling some given distance downstream of the nozzle exit. Presented in Figures 9-40 thru 9-42 are comparison data for the two elevon configurations, at three nondimensionalized chord stations of 10t, 20t, and 50t. Improvements were noted at all three chord stations as well as along the entire span.

Flow visualization pictures for purposes of illustrating flow angularity were made at 15 PSIG for both the Proto #1 and improved elevon. Figures 9-43 and 9-44 present the improvements that were realized, especially at the inboard corner, which can be attributed to the improved turning vanes located in the inboard dedicated nozzle. To further quantify the flow angularity, oil flow techniques were used to obtain the following flow angles along the span.

NOZZ STA (IN)	10	16	26	34	Midspan
FLOW ANGLE (DEG)	7.5	7	3	2	0

Figure 9-31. Nozzle Gap Measurements

Figure 9-32. Gap at Nozzle Lip After Pressurization
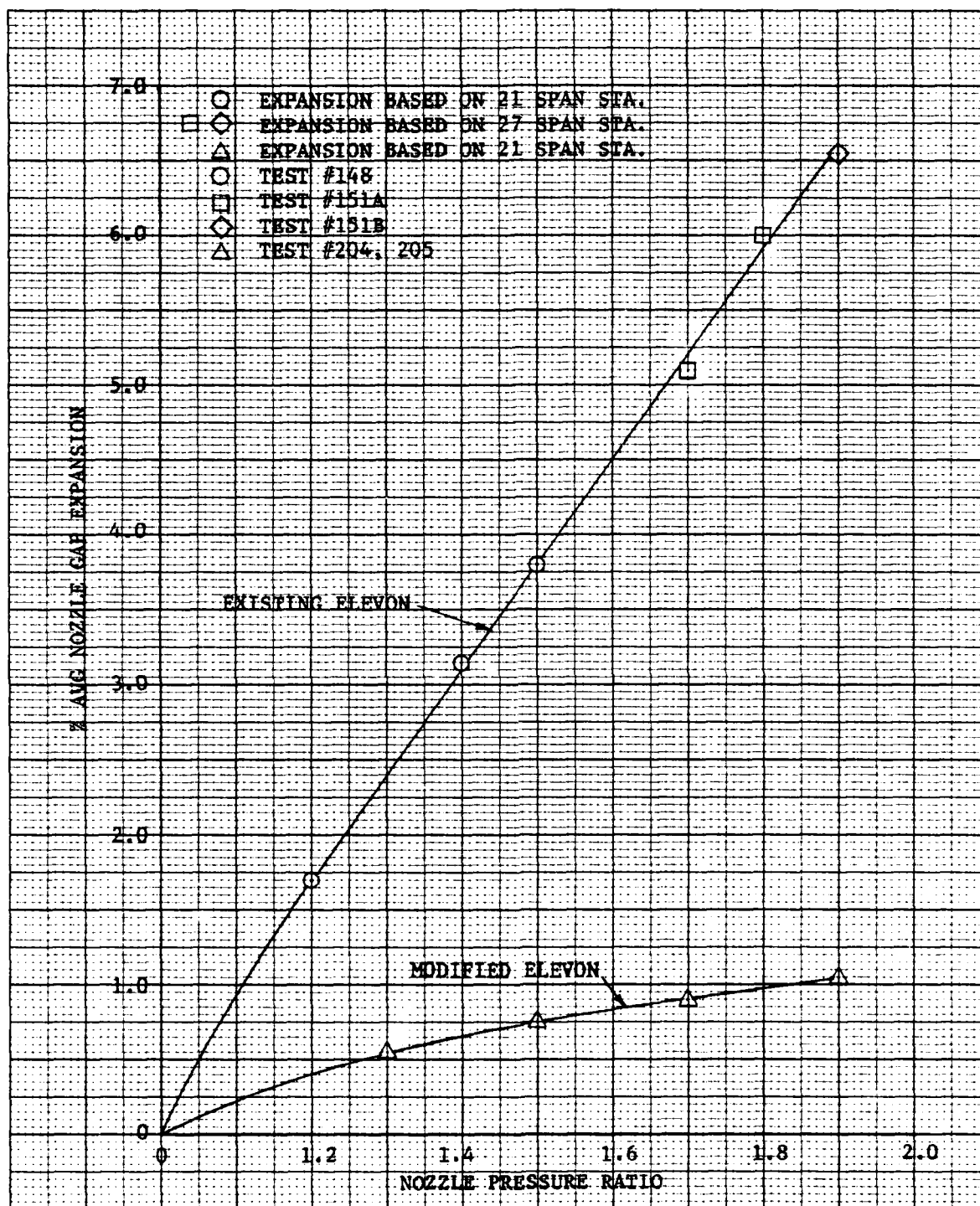
Figure 9-33.   Nozzle Gap Expansion
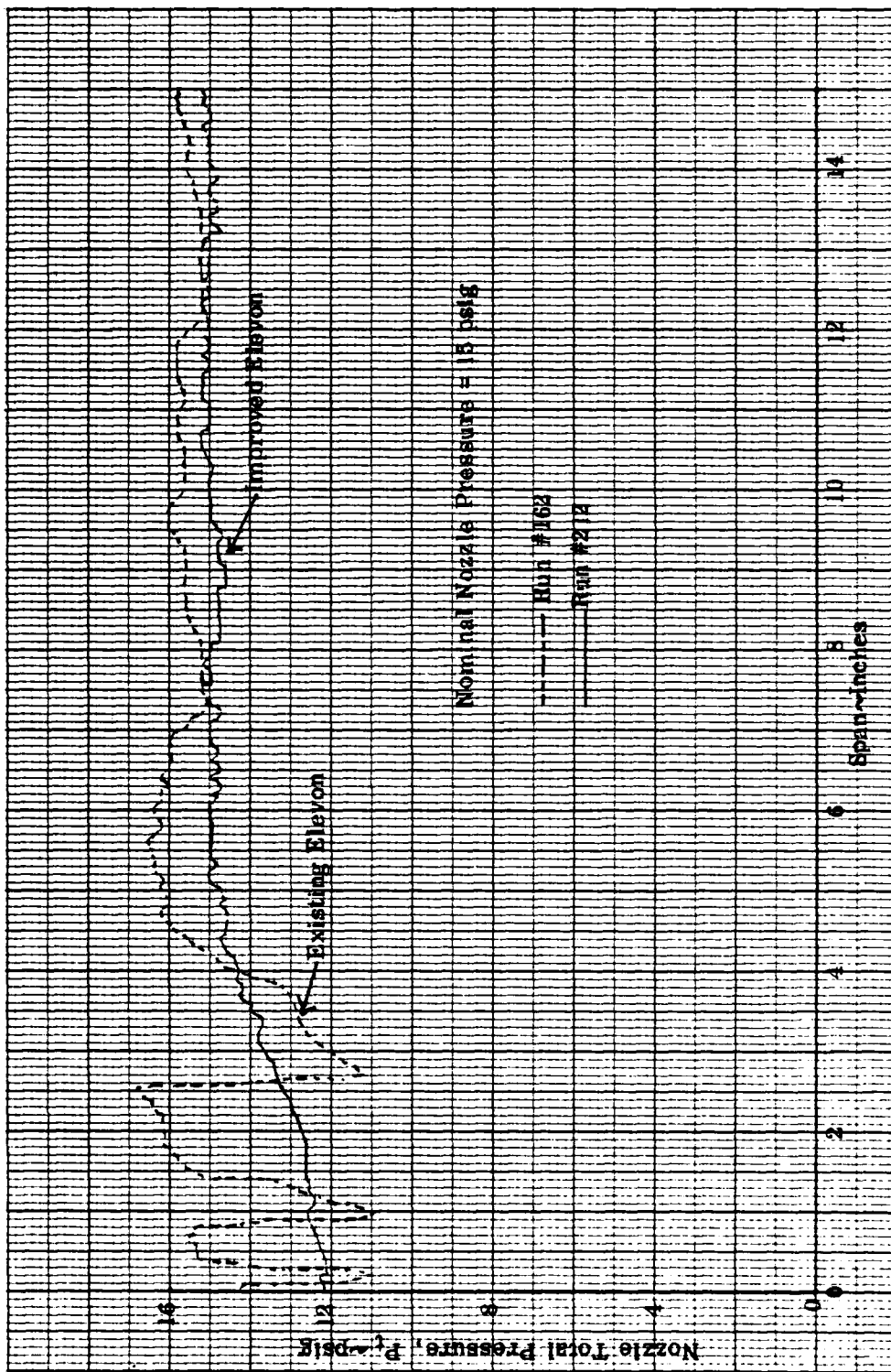
Figure 9-34. Primary Nozzle Survey Comparison - Inboard

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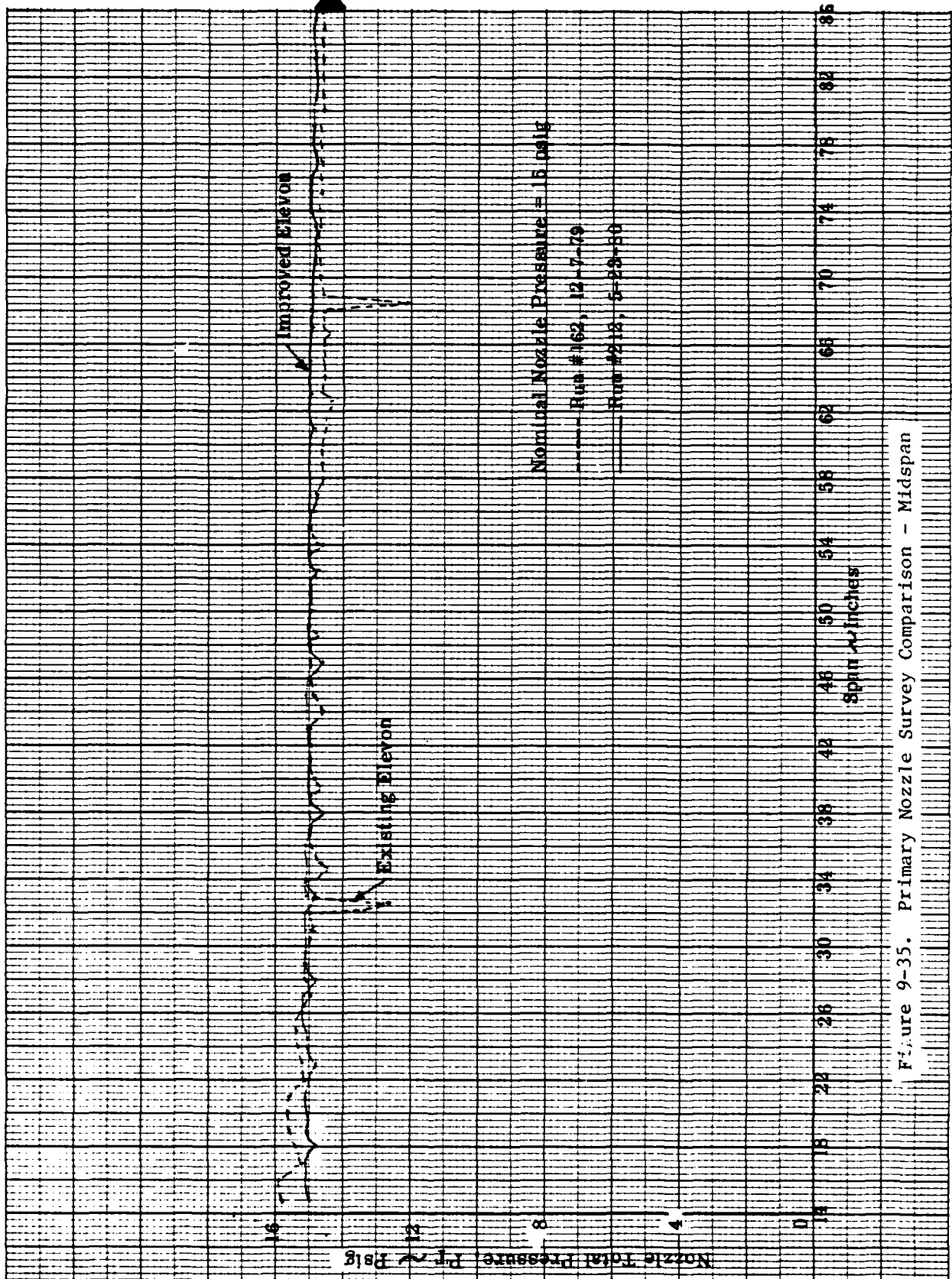
Figure 9-35. Primary Nozzle Survey Comparison – Midspan
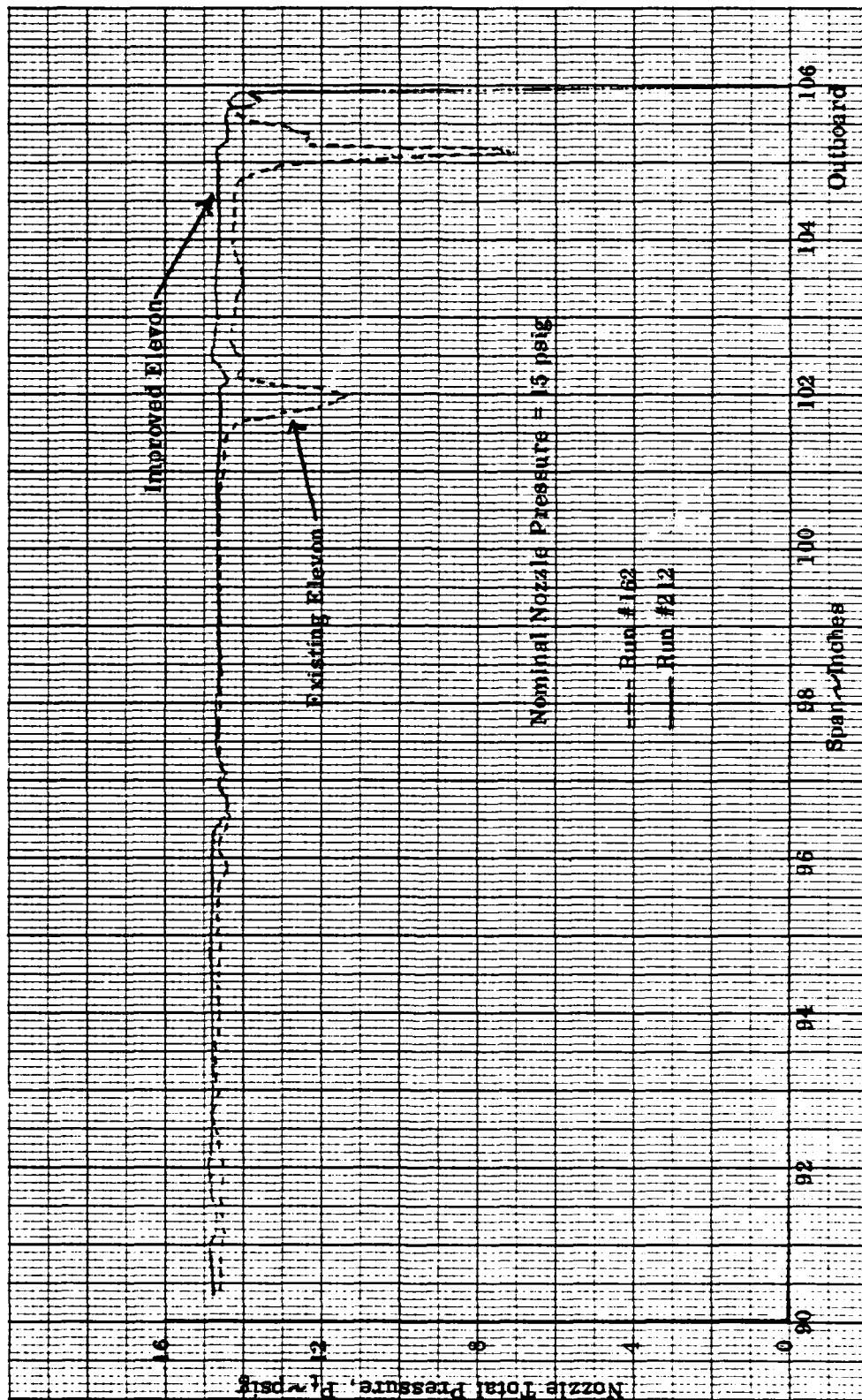
Figure 9-36. Primary Nozzle Survey Comparison – Outboard
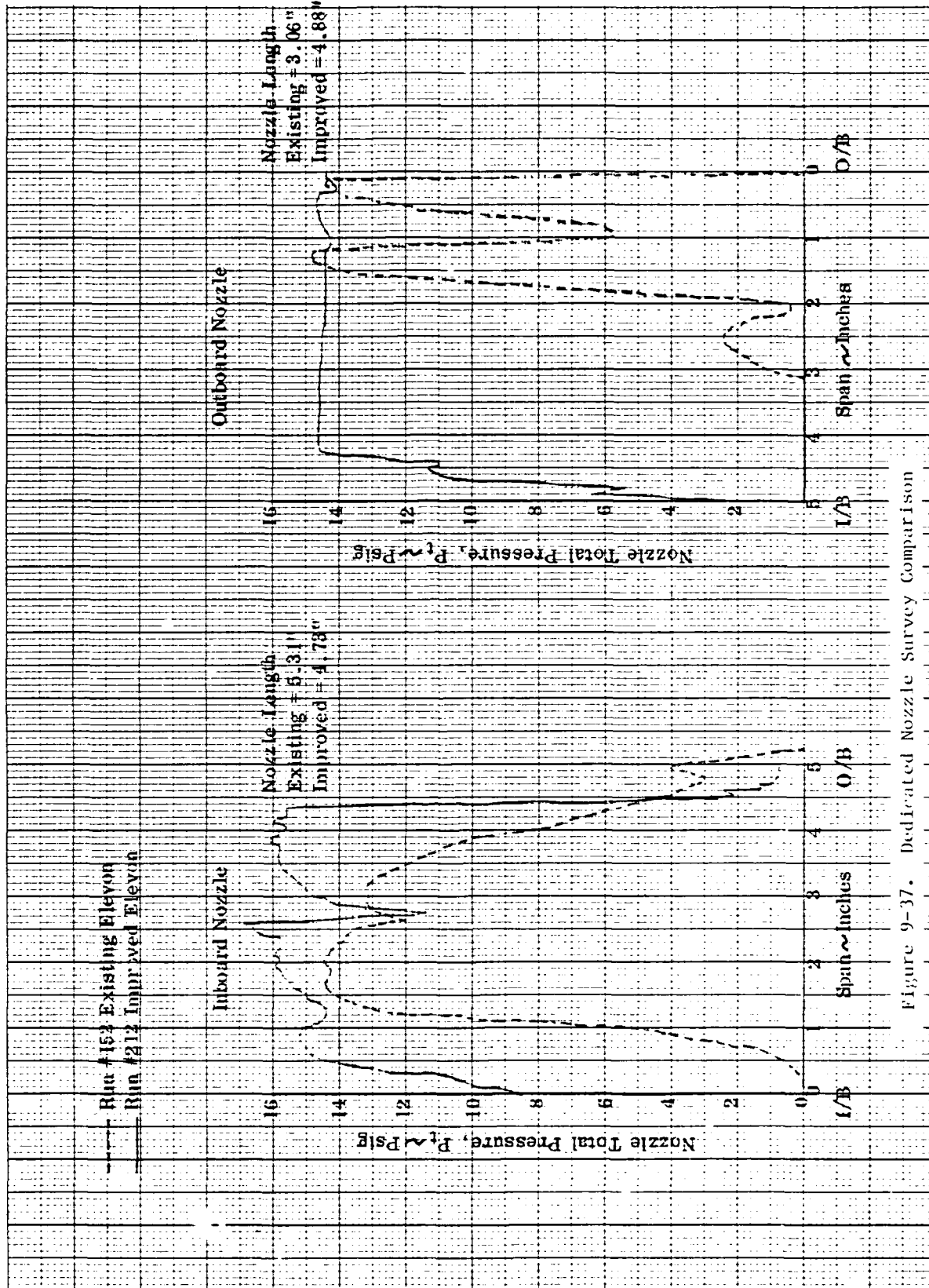
Figure 9-37. Dedicated Nozzle Survey Comparison
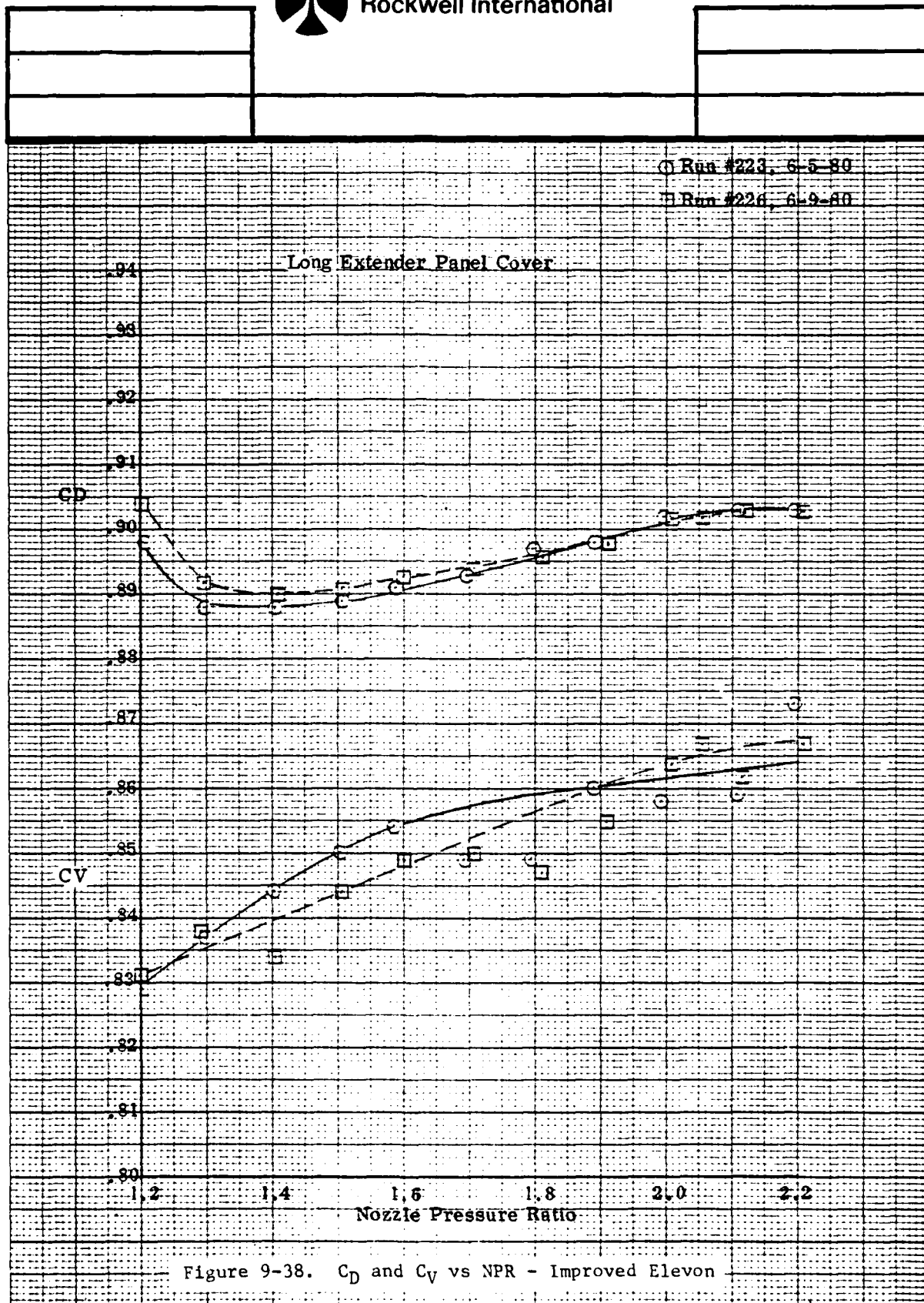
Figure 9-38. C_D and C_V vs NPR - Improved Elevon

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Figure 9-39. C_D and C_V vs NPR - Existing Elevon
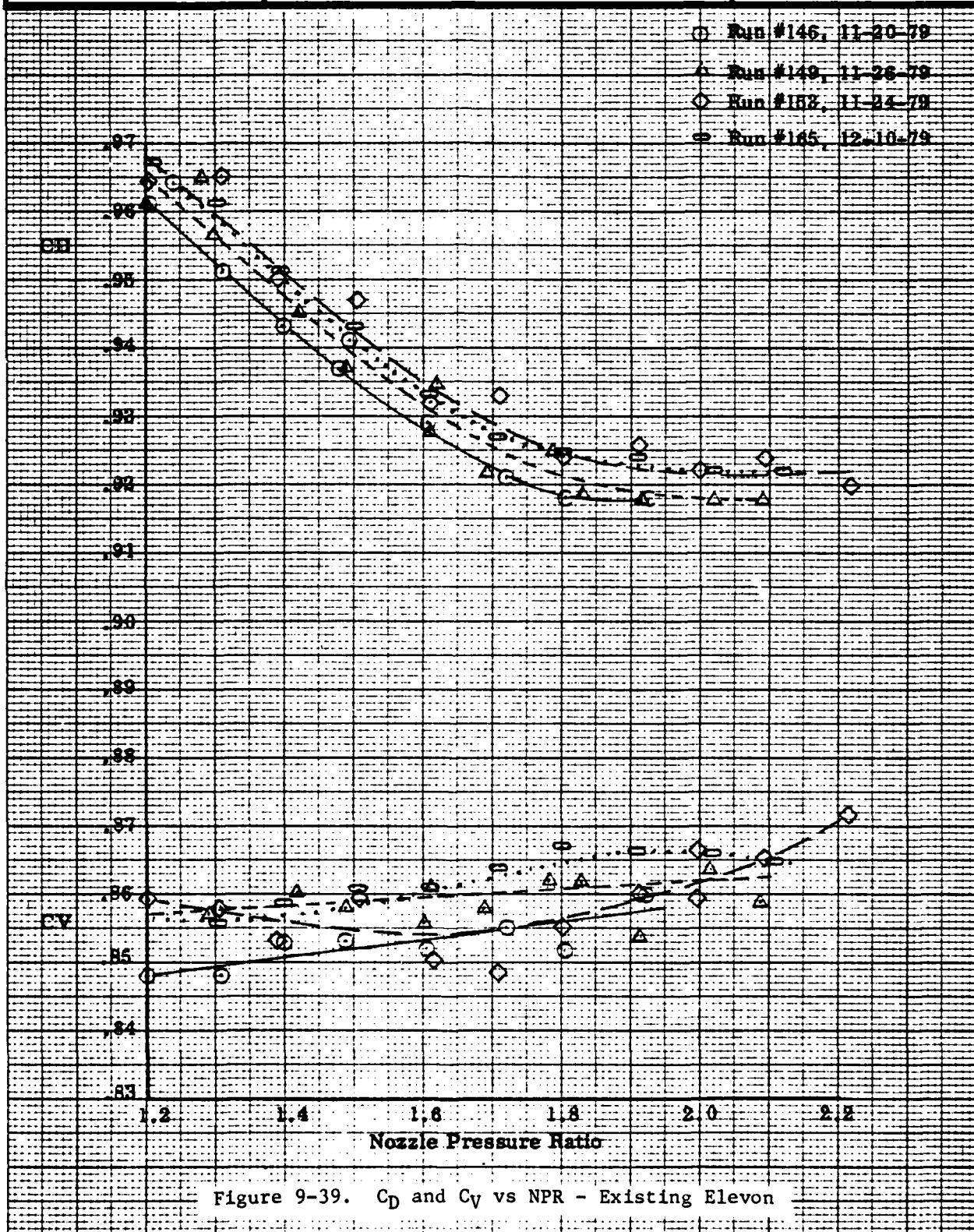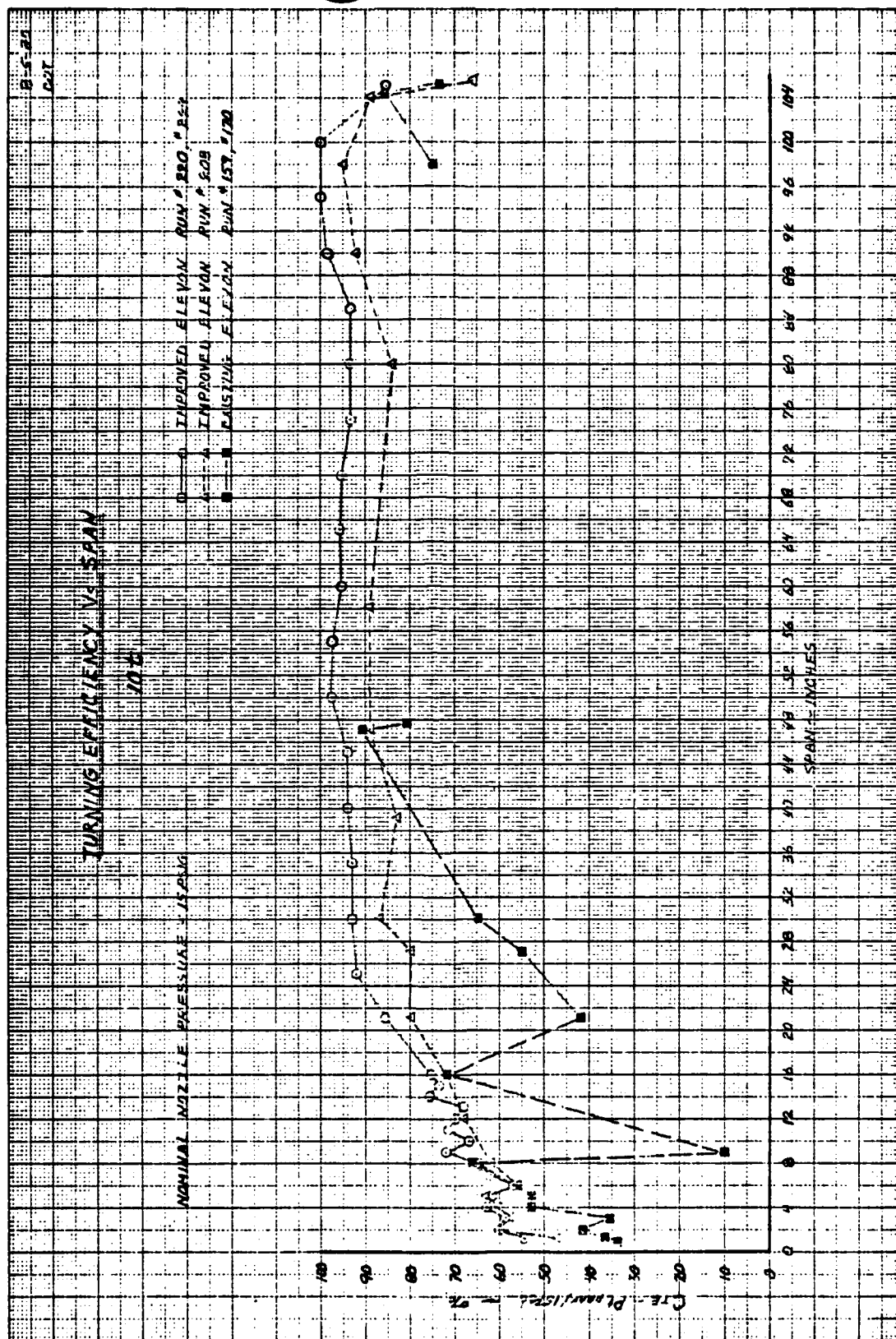
Figure 9-40. Turning Efficiency vs. Span – 10t

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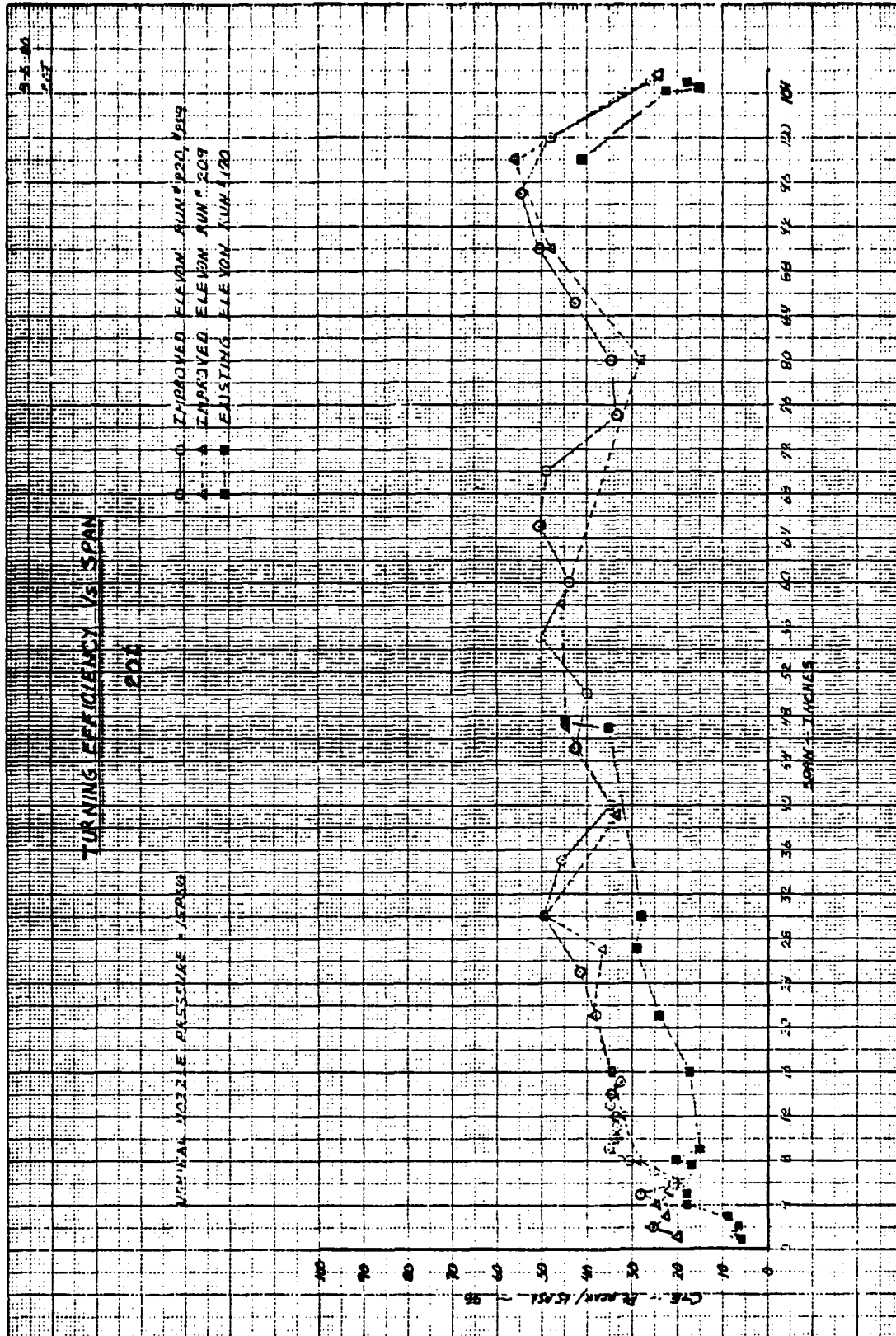
Figure 9-41. Turning Efficiency vs. Span – 20t
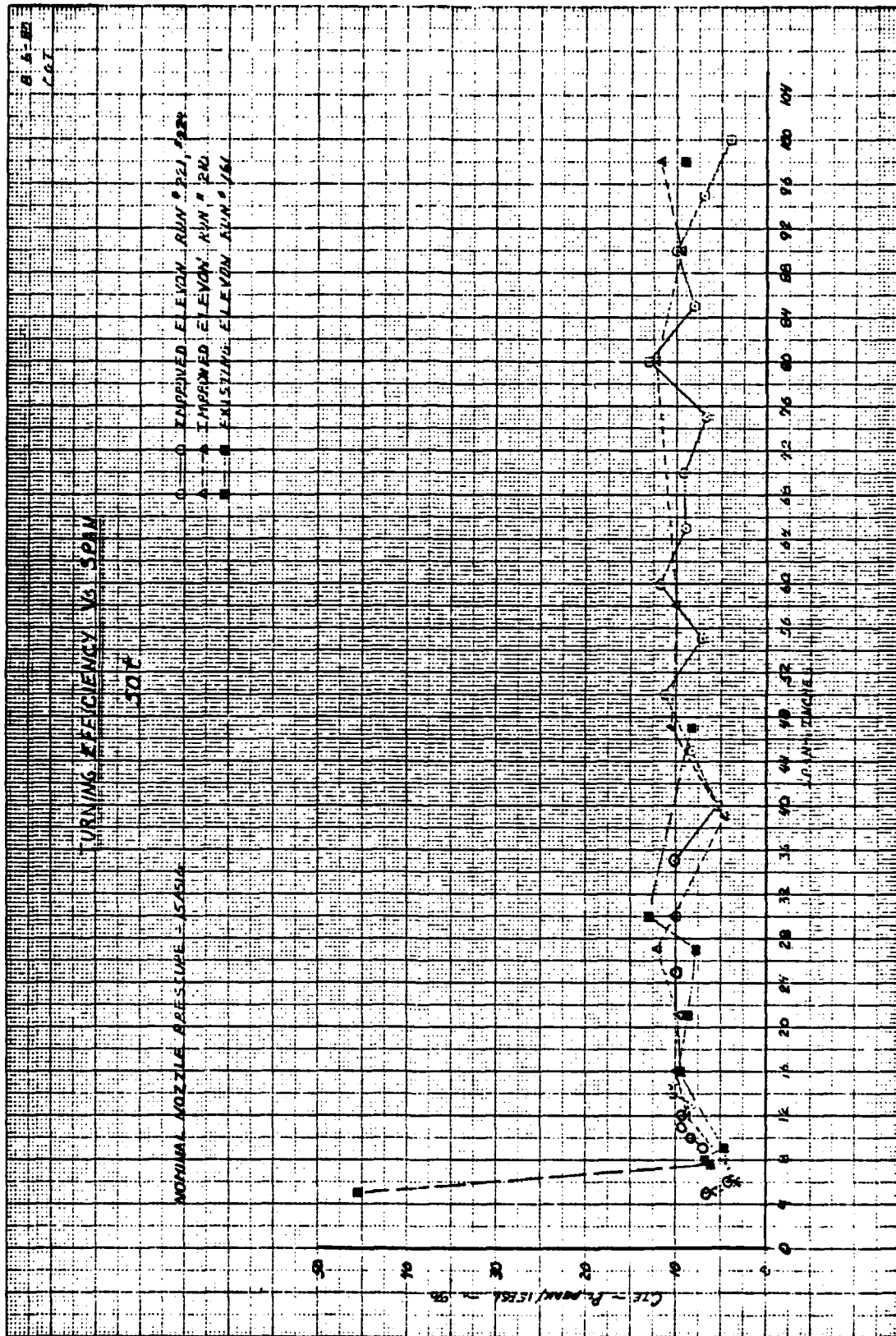
Figure 9-42. Turning Efficiency vs. Span – 50t

IMPROVED ELEVON

Noz Sta 1"

Noz Sta 4"

EXISTING ELEVON

Noz Sta 1"

Noz Sta 4"
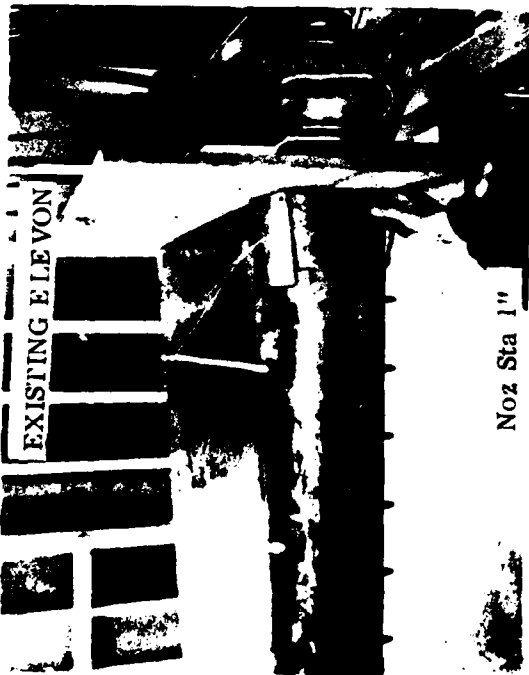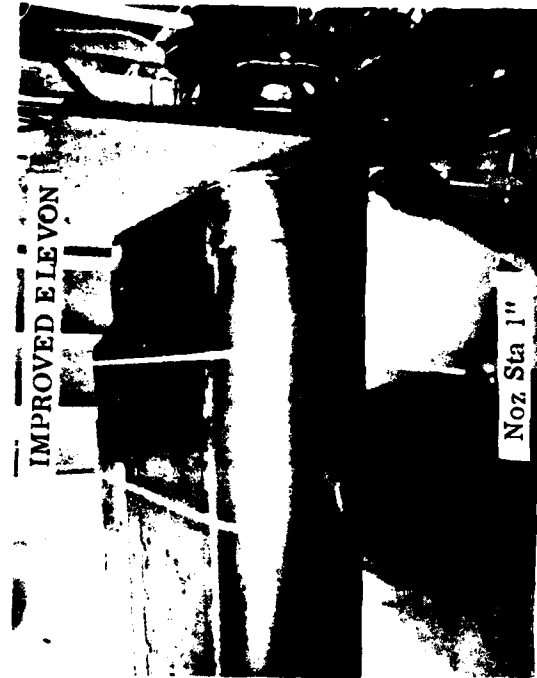
Figure 9-43. Flow Visualization Comparison at 1 and 4 Inches
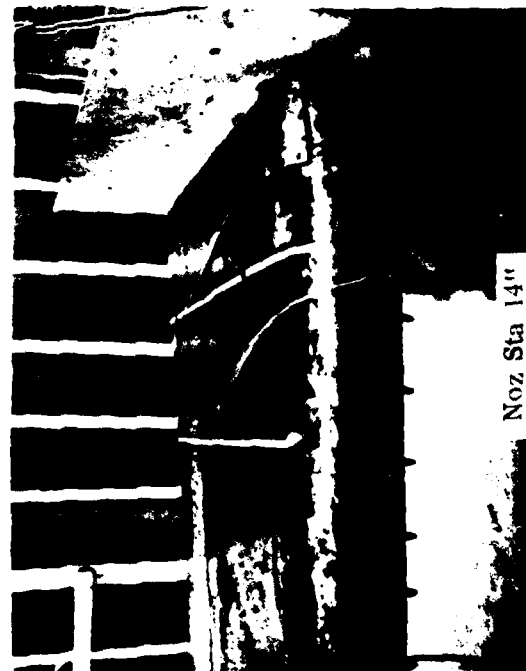
Figure 9-44. Flow Visualization Comparison at 8 and 14 Inches

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9.4   SUMMARY

Flow characteristics of the augmenter surfaces from the left hand wing of
the Proto 1 XFV-12A were evaluated in cold flow tests. Spanwise nozzle
total pressure distributions, jet profiles, nozzle coefficients and flow
visualization were obtained for the wing diffuser, elevon, and inboard and
outboard endwall nozzles. In addition, leakage and nozzle area growth
were investigated for the diffuser and elevon. The data obtained was
considered to be satisfactory for comparison with similar data of the
improved augmenter surfaces.

Also presented are the improvements in overall flow quality and nozzle gap
expansion control obtained on the elevon which had several design
improvements made to it which included: the original Coanda shape log
spiral changed from $K = 1.0$ to $K = 1.3$, a three vane cascade and a bullet
fairing added to the inboard dedicated nozzle, increased nozzle lip
convergence from 0° to 7.5°, improved Coanda surface smoothness by
eliminating splice joints, and removal of a primary nozzle turning vane
which became unnecessary as a result of an improved spacing relationship
between the feed and nozzle exit plane which resulted from the change in
Coanda K factor.

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10.0 TEST FACILITIES

10.1 GENERAL

In order to pursue the development rationale discussed in Section 3.1, it was necessary to establish test facilities capable of performing accurate performance measurements and diagnostic investigations of both model scale and full scale thrust augmenter systems and their associated components. For this purɂ ɂ, three major test facilities were utilized routinely during the XFV-12A TAW technology program. Table 10-1 presents a brief outline of each facility, its utilization, and specific operating characteristics applicable ʳo thrust augmenter research programs.

Table 10-1 Major Test Facilities

TEST FACILITY	HARDWARE EVALUATED	FACILITY CHARACTERISTICS
Thermodynamics Laboratory	Model Scale Augmenters Model Scale Components Full Scale Components	Compressor supplied cold flow models operating at aircraft nozzle pressure ratios (NPR's). Full scale components limited to NPR's $\leq$ 1.9:1
WPAFB Test Facility	Full Scale Components Full Scale Segment Models	Compressor supplied cold flow tests performed at NPR simulating aircrat operating conditions.
Whirl Rig	Full Scale Augmenters Full Scale Components	F401 engine supplied flow simulates aircraft nozzle exit pressure and temperature characteristics

The remainder of this section will present detailed information concerning each of the facilities and their specific applications.

10.2 WHIRL RIG TEST FACILITY

Full scale augmenter and associated isolated component tests were conducted at the contractor's whirl rig test facility. This facility which is located at Rockwell International, NAAD Columbus Plant, was specially designed and fabricated for this application. The facility is basically composed of a P&WA F401 engine with inlet bellmouth and exhaust diverter, a 100-foot section of boom and ducting required to deliver a portion of the engine air flow to the item under test (ɛee Figures 10-1 to 10-4). The facility was originally designed to allow the test article to rotate around the boom center pylon in order to demonstrate hover and control aspects of the TAW concept. Typically, however, the facility was utilized for diagnostic or performance verification tests. Investigations of this type utilized a force measuring system composed of load cells
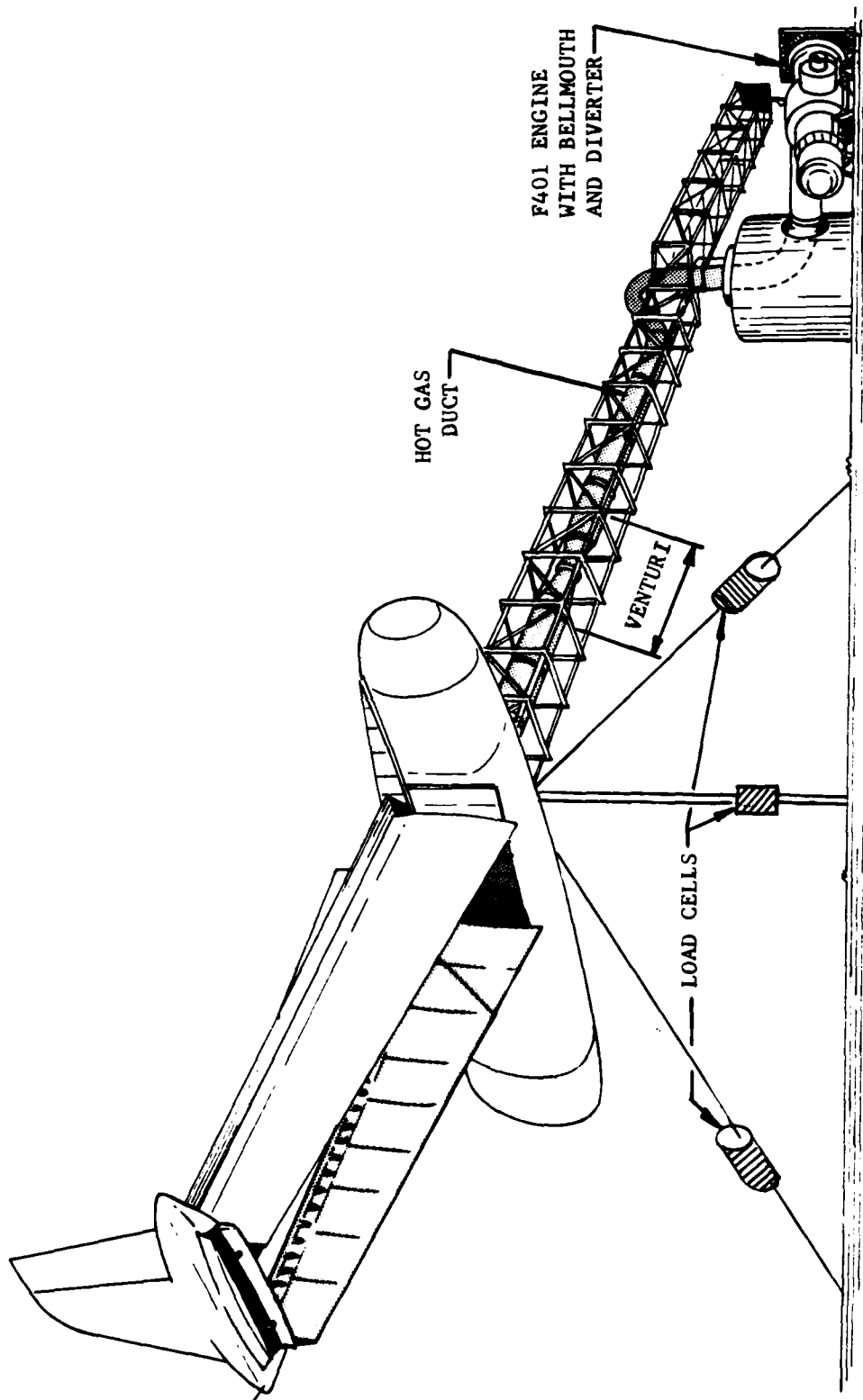
Figure 10-1. Schematic - Whirl Rig with Model Installed
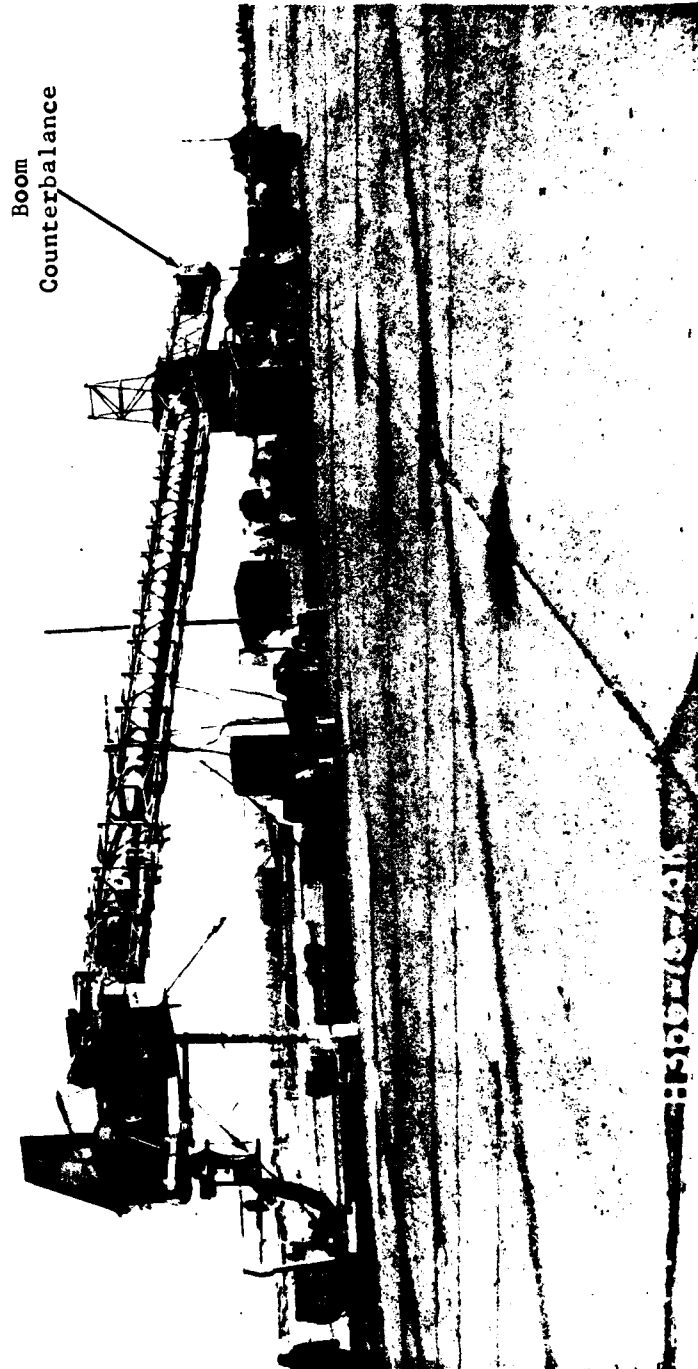
Figure 10-2. F401 Whirl Rig with Wing Model Installed

PLENUM

DUMP DOOR
FOR EXCESS
ENGINE
HOT GAS

ENGINE

ENGINE
STAND

HOT GAS
DUCT

BOOM

FOD SCREEN

11/10/78
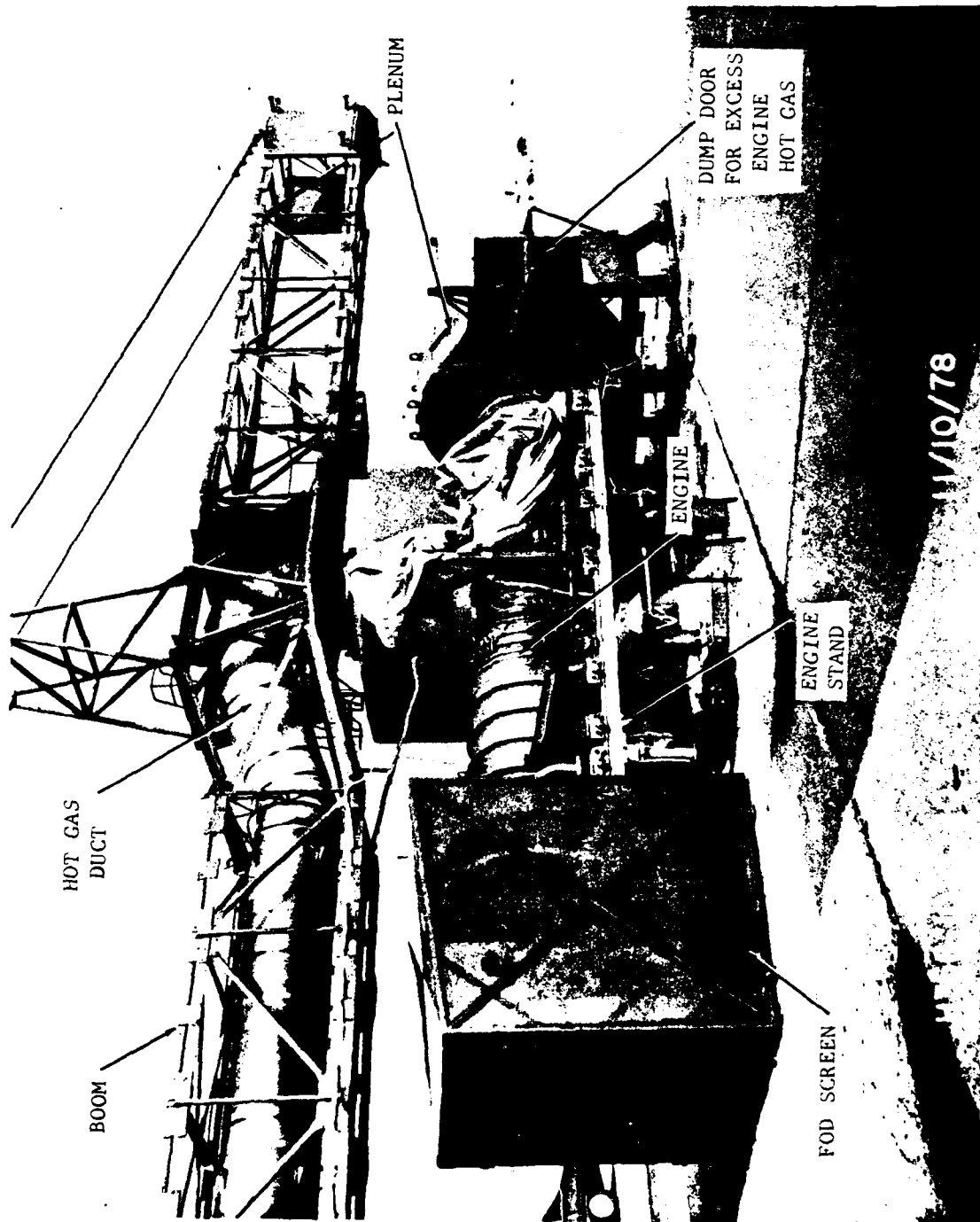
Figure 10-3. F401 Engine Installation

PYLON
BOOM SUPPORT

FUEL TANK
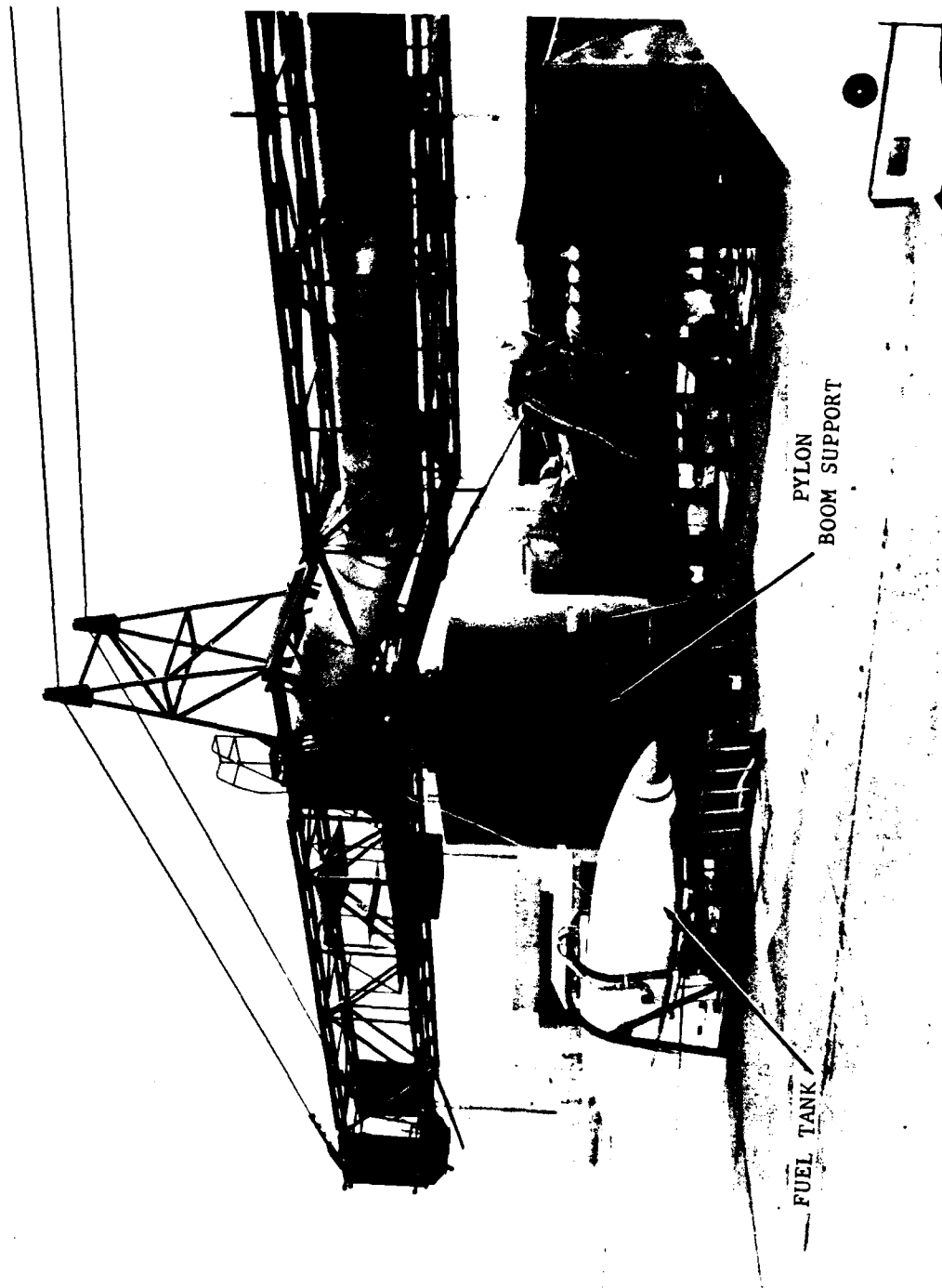
Figure 10-4. F401 Engine - Installation

located at the end of the boom to obtain test article lift and drag data. Measurements of engine airflow to the test article were obtained from a calibrated venturi located in the upstream boom structure or in the case of low flow conditions a calibrated orifice plate was substituted for the venturi. Associated instrumentation (pressure and temperature sensing devices) was located as necessary in order to determine the desired performance.

10.2.1 Instrumentation and Data Acquisition

The instrumentation utilized during the whirl rig test program is essentially the same as that utilized in model scale laboratory tests. Pressure and temperature probes, load cells, traversing probes, amplifiers, and scanivalves are employed to isolate model performance characteristics. Engine operational characteristics and health are monitored utilizing fuel flow electronics, rpm signal box, vibration accelerometers, bellmouth pressure and temperature sensors, and various other diagnostic instrumentation. The major elements of the data acquisition and display system include the block house control room, instrumentation van, and test support equipment (see Figures 10-5 to 10-7).

A typical list of data requirements during a full scale augmenter test program is presented in Table 10-2. Subsequent to acquisition, test data is then reduced and analyzed by the engineering staff to determine the performance characteristics of the test article.

10.2.2 Engine Instrumentation

An instrumented bellmouth with a conical FOD screen is attached to the engine. The engine is equipped with a P&WA instrumentation kit; however, some of the secondary fuel and oil system sensors are not connected for data acquisition. Sensors required for engine operation, trim, health monitoring and performance level determination are utilized.

Primary engine parameters such as FTIT, N1, N2, CIVV, RCVV, etc., are displayed on indicators at the engine operators console. Warning and status penalties are also provided for the engine facility power and fuel supply systems. Facility hydraulic system status and the test model surface positions are displayed at the flight control operations console. Separate DVMs are provided for each position readout. Control of the three surfaces are by individual hydraulic actuators.

Mercury and water manometers are used to set up the engine for specific test conditions and determine fan pressure ratio and airflow at the engine bellmouth. Manometers are also used to display either the venturi or orifice pressures. Pressure levels at the simulated diverter entrance and in each augmenter nozzle exit are also monitored. Parameters connected to the manometers are also transmitted to the telemetry receiving station. Readouts from the load cells and resolved lift and drag forces are recorded on a digital multipoint recorder at selected time intervals.
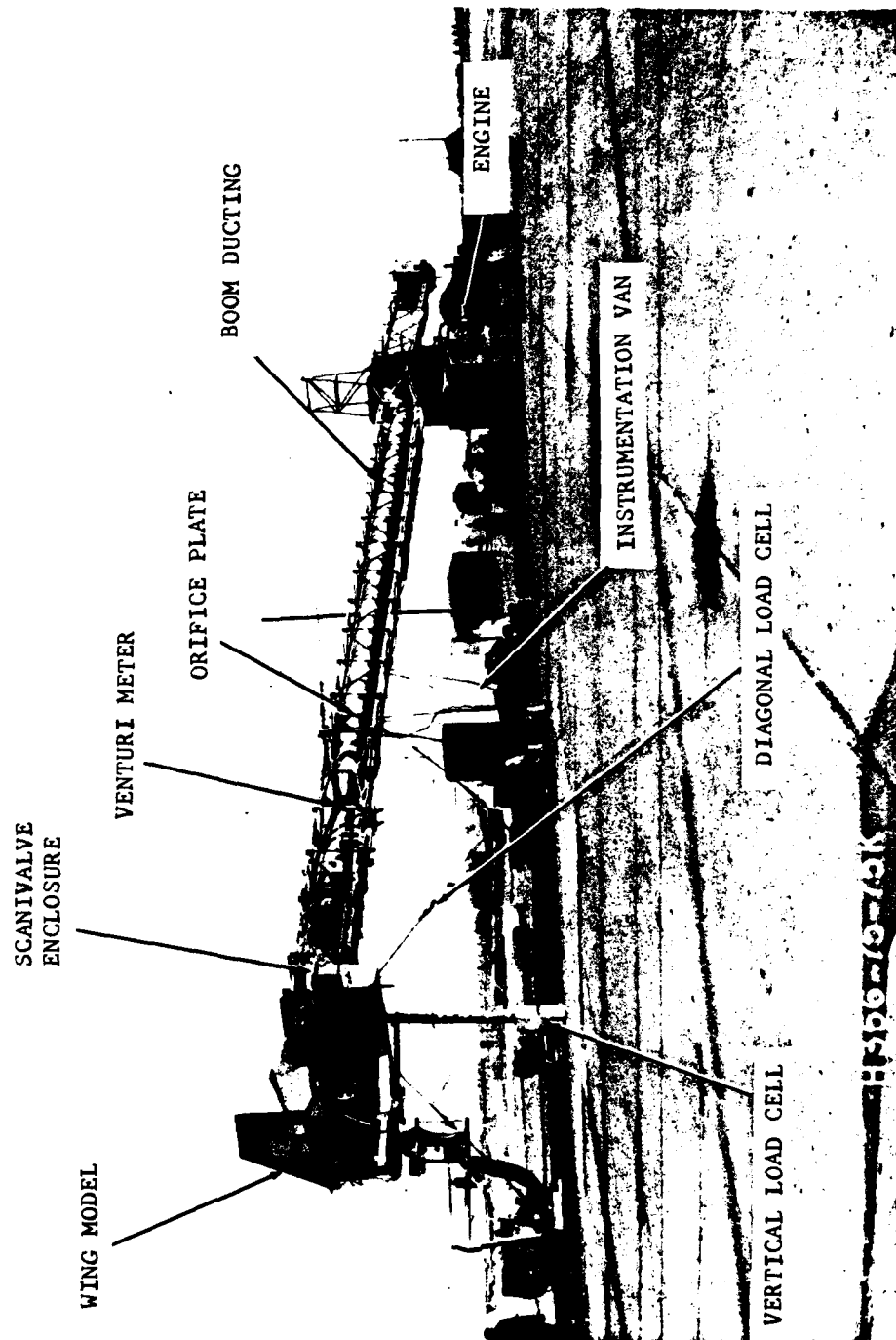
Figure 10-5. Whirl Rig - Wing Model Instrumentation Arrangement
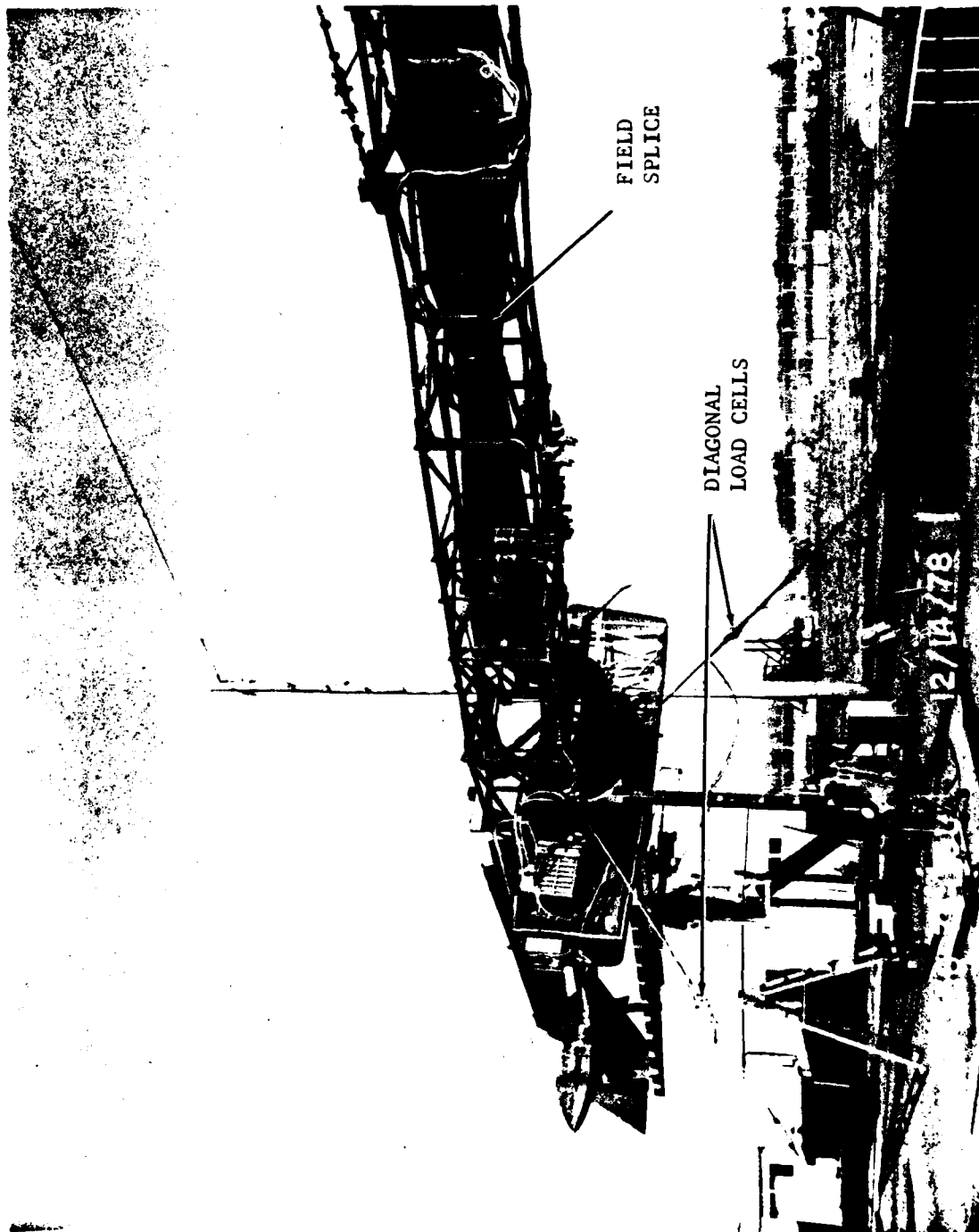
Figure 10-6. Outboard End of Boom with Model Installed

FIELD SPLICE

DIAGONAL LOAD CELLS
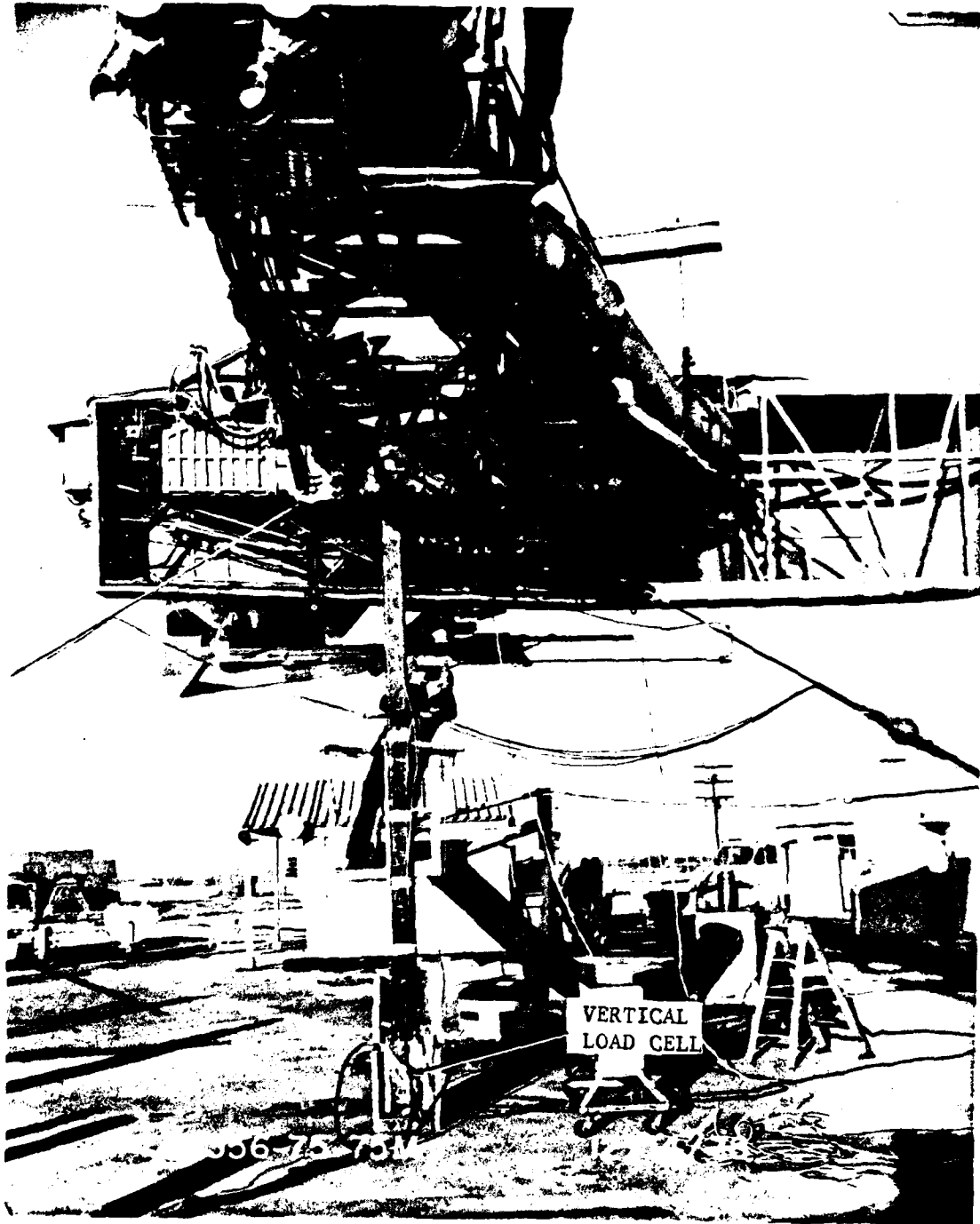
Figure 10-7. Outboard End of Boom with Model Installed

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Table 10-2 Whirl Rig Instrumentation Parameters

No. Parameters

Engine

Bellmouth Pressures and Temperatures	8
Control Input Positions	4
Gas Temperature and Pressures	6
Speed and Fuel Flow	3
Fuel and Oil Pressures and Temperatures	3
Vibration (Not Recorded)	3
	27

Boom Ducting

Venturi Static Pressures and Temperatures	4
Orifice Static Pressures	2
Boom Discharge Total Pressure	1
	7

Model

Interface Total Pressure and Temperatures	5
Surface Positions	3
Center Ejector Nozzle Exit Total Pressures	20
Forward Diffuser and Elevon Nozzle Exit Total Pressures	12
Augmenter Endwall Nozzle Exit Total Pressures	4
Augmenter Nozzle Expansion (5 each nozzle)	15
Center Ejector Surface Static Pressures	17
Forward Diffuser Surface Static Pressures	6
Augmenter Throat Static Pressures	10
Scanivalve Signal and Control	5
	97

Traversing Probe

Augmenter Exit and Throat Total and Static Pressures	6
Augmenter Total Temperatures	2
Center Ejector Jet Profile Pressures	2
Probe Span and Chord Position	2
	12

Facility

Loads	3
Wind Speed and Direction	5
	8

TOTAL   151

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10.2.3   Instrumentation Van

The van shown in Figure 10-5 is a multipurpose test installation which is
used during calibrations, pre-test checkout and actual test runs. The van
is located adjacent to the whirl rig boom and contained the primary data
acqusition components which are employed during aircraft testing.
Transducers are housed for the model and boom parameters to eliminate the
need for separate enclosures. The PCM decom and a strip chart recorder
are used during pre-test checkout and surveillance of selected data
channels during the test runs. Engine vibration sensor outputs are
displayed on the P&WA engine vibration monitor rack. Control of the
traversing probe during span and chord positioning is also accomplished
from the van. The engine trim box is temporarily installed during initial
checkout runs of the engine.

10.2.4   Model Instrumentation

Measurements obtained to isolate model performance includes the
measurements taken on the model and are basically as follows: feed duct
interface pressures, nozzle total pressures, nozzle expansion, surface
positions, center ejector forward and aft surface static pressures and
augmenter throat static pressures, mass flow rates and temperatures, lift
and drag, and traversing rake surveys.

Coanda component nozzle exit total pressures are normally recorded at six
span stations. Since the center ejector profiles typically show
significant variation, twenty probes are employed to investigate the
pressure gradients and determine average pressure levels. Probes are also
installed in the inboard and outboard endwall blowing nozzles.
Figures 10-8 and 10-9 show typical forward diffuser and center ejector
probe placement. Each pressure is sampled at 2.5 second intervals
utilizing a scanivalve. Nozzle average pressure data is used to compute
component primary isentropic nozzle exit velocity.

Coanda and center ejector nozzle gap deflections are measured using a
linear potentiometer attached to the nozzle and located outside the
primary jet. The rod end of the potentiometer is coupled to the flap
surface by a ceramic rod. Prior to installation, the potentiometers are
calibrated to verify linearity over the operating range. This expansion
data is used to determine individual component flow coefficients. This
information is also used to isolate the effects of pressure and
temperature on nozzle covergence.

A 20-inch diameter (throat) venturi is installed approximately 25 feet
from the end of the boom to measure gas flow during full augmenter tests.
The venturi meter discharge coefficient is determined over the test
operating range using a calibrated flow nozzle prior to testing. Sharp
edge orifice meters of 12.75 in. and 17.6 in. diameter are installed
upstream of the venturi for component testing. This provides a
differential pressure readout equivalent to that of the venturi meter at
maximum flow. Total pressure and temperature probes are installed
upstream of the aircraft ducting at the 32-inch duct discharge and
simulated diverter entrances.
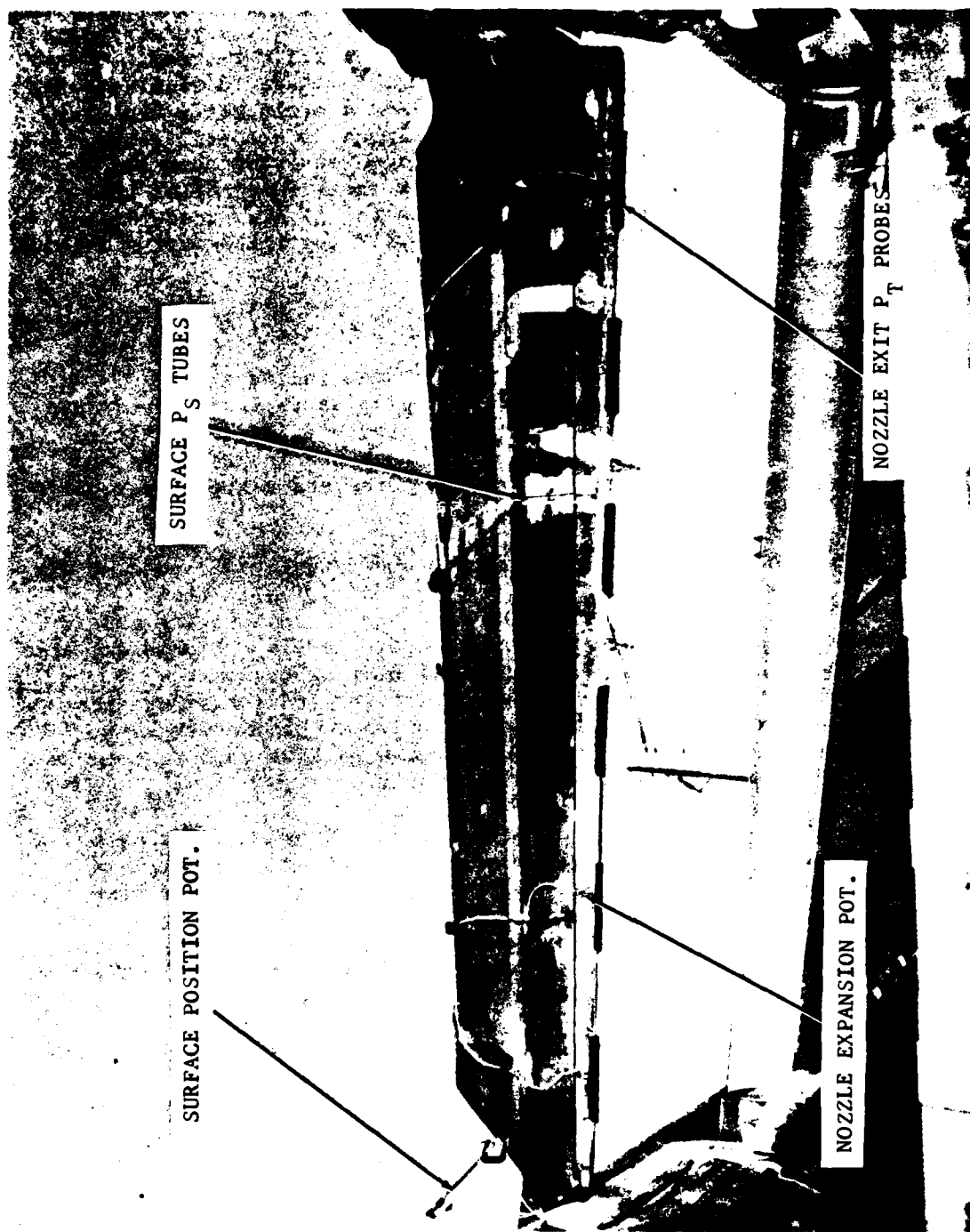
Figure 10-8. Wing Center Ejector Instrumentation
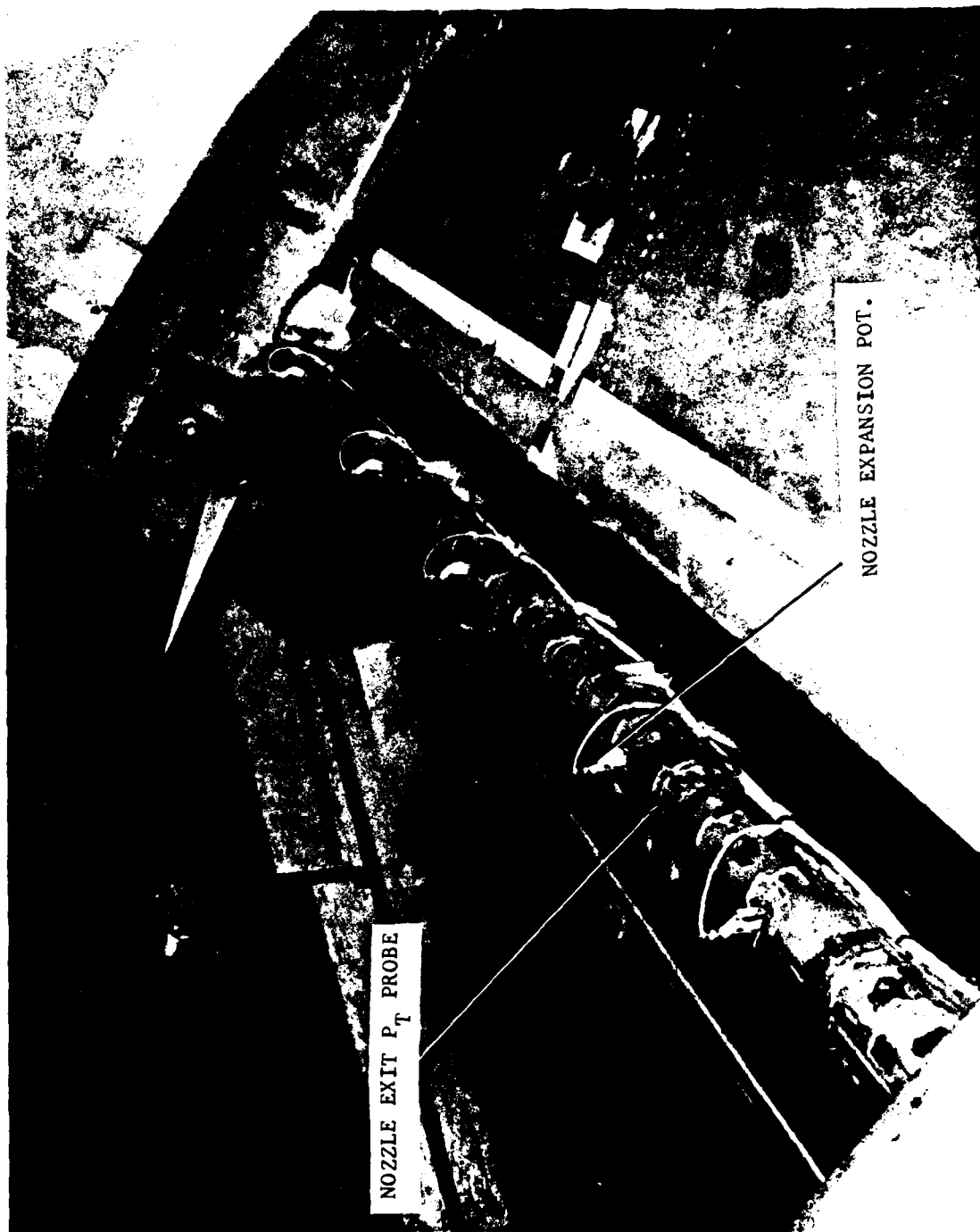
Figure 10-9. Wing Forward Diffuser Instrumentation

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Venturi gas temperature and throat surface temperature are recorded and appropriately used in the computation of mass flow. Leakage flow downstream of the orifice meters and venturi are determined during a special test wherein an equivalent leakage area is computed and considered in the measured net flow to the test article.

Center ejector surface static pressure tubes are often installed at mid-span on the forward and aft surface to determine down loads during full augmenter testing (see Figure 10-8). The surface taps and surface position potentiometer installation is shown in Figure 10-9. Augmenter forward and aft throat static probes (10) are located at the same aircraft test positions (see Figure 10-10). These static probes provide the data necessary to compute augmenter throat secondary velocity, an empirical parameter routinely used to quickly determine relative levels of augmenter performance.

The lift and drag forces produced by the augmenter are measured by a three-component load cell system shown in Figure 10-6 and 10-7. These forces are corrected to the estimated augmenter lift center location for evaluation of augmentation ratio and component thrust coefficient. Transducer incorporated 10,000 pound load cells previously used at Langley are installed in the system. Prior to installing the units a full range calibration is performed with a dead weight tester. Each load cell has two strain gage circuits to provide a dual signal output. One bridge is used to record by PCM or magnetic tape for use in post test data reduction. The second bridge is connected to a Digitrend 210 digital multipoint recorder located in the control room. This unit provides a digital display and printout in engineering units. Each individual load cell readout as well as a summation of lift and drag are printed out using manual select at two samples/sec.

The force measurement system consists of a vertical post held in place by diagonal tie rods. Tension preloading of the diagonal tie rods is used to produce a compression preloading of the vertical post and maintain a slack free redundant force system. Counterbalance is used to adjust the boom tare weight to the desired level.

The vertical load cell output is amplified in order to scale the 1024 PCM counts to 2000 pounds (maximum load cell change from zero balance). The diagonal load cell outputs are amplified to accommodate a load change of 4000 pounds. Even though the load cells are temperature-compensated over a range of 0 to 150°F, heater elements are wrapped around each unit to maintain a near constant temperature at 90° $\pm$15°F.

Utilizing the information obtained from the load cell system and the primary flow characteristics determined by the venturi, pressure and temperature instrumentation previously discussed enables the engineering personnel to compute total performance indicators such as augmentation ratio (ϕ), isolated component velocity coefficients (C_V), and nozzle flow coefficients (C_D).

FWD & AFT @
2" FROM E/W

FWD & AFT @
20" FROM E/W

ELEVON

FWD & AFT
@ MIDSPAN

H-356-75-76 M   12-22-78

Figure 10-10. Wing Augmenter Throat P_S Probes

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10.2.5   Traversing Probe

The traversing probe assembly is used to isolate the relative test article flow quality and determine localized augmenter performance. This assembly consists of a carriage moving along a chordwise rail supporting pitot-static pressure probes and a thermocouple temperature sensor (see Figure 10-11). The chordwise rail is mounted between fore and aft span rails and is movable along them. Both motions are motor-driven and controlled from an isolated position. Position sensors are incorporated in each motor gearbox to provide instrumentation chordwise and spanwise position information. The height of the probes is adjustable by sliding the sensors through clamping supports thus providing capability to survey various x/t locations downstream of the primary nozzle exit (see Figure 10-12). Pressure and temperature data are transmitted via telemetry and recorded on X-Y strip charts. This data is then evaluated to provide information on local thrust, mass flow, velocity, temperature and relative flow quality of the augmenter or component under test.

10.3   THERODYNAMICS LABORATORY

Rockwell International, NAAD-Columbus' Thermodynamics Laboratory is an environmental test facility which provides support for conducting research and development in the fields of internal fluid flow, heat transfer, and thermodynamic properties of materials and systems. This facility is located in Building 271 and has a total floor space of over 27,500 square feet. The major equipment and assets in the Thermo Lab, which provided routine support of the XFV-12A V/STOL program, included the high pressure compressor air process system, hazard proof test chambers, and data system and support instrumentation for the control and measurement of the parameters specified. Specifics of the individual test requirements and programs are discussed in the apparatus and test result sections of this report.

The high pressure compressor air process system is composed of a compressor, heater, control, and support distribution systems. A schematic layout of the system is shown in Figure 10-13. The following paragraphs provide a brief description of the major system components and their respective operating ranges.

10.3.1   Air Compressor

The Ingersoll Rand Compressor is a ten-stage turbo compressor operating at 9200 RPM and currently delivering airflow at 13 pounds/second at 200 psig. Figure 10-14 presents available airflow delivery in the form of compressor flow rate versus compressor discharge pressure specified and currently available levels.

10.3.2   Control and Distribution System

The control and distribution system is the responsibility of the laboratory operating staff. Predetermined pressure and flow requirements are set for the compressor at the main control panel. Second order or
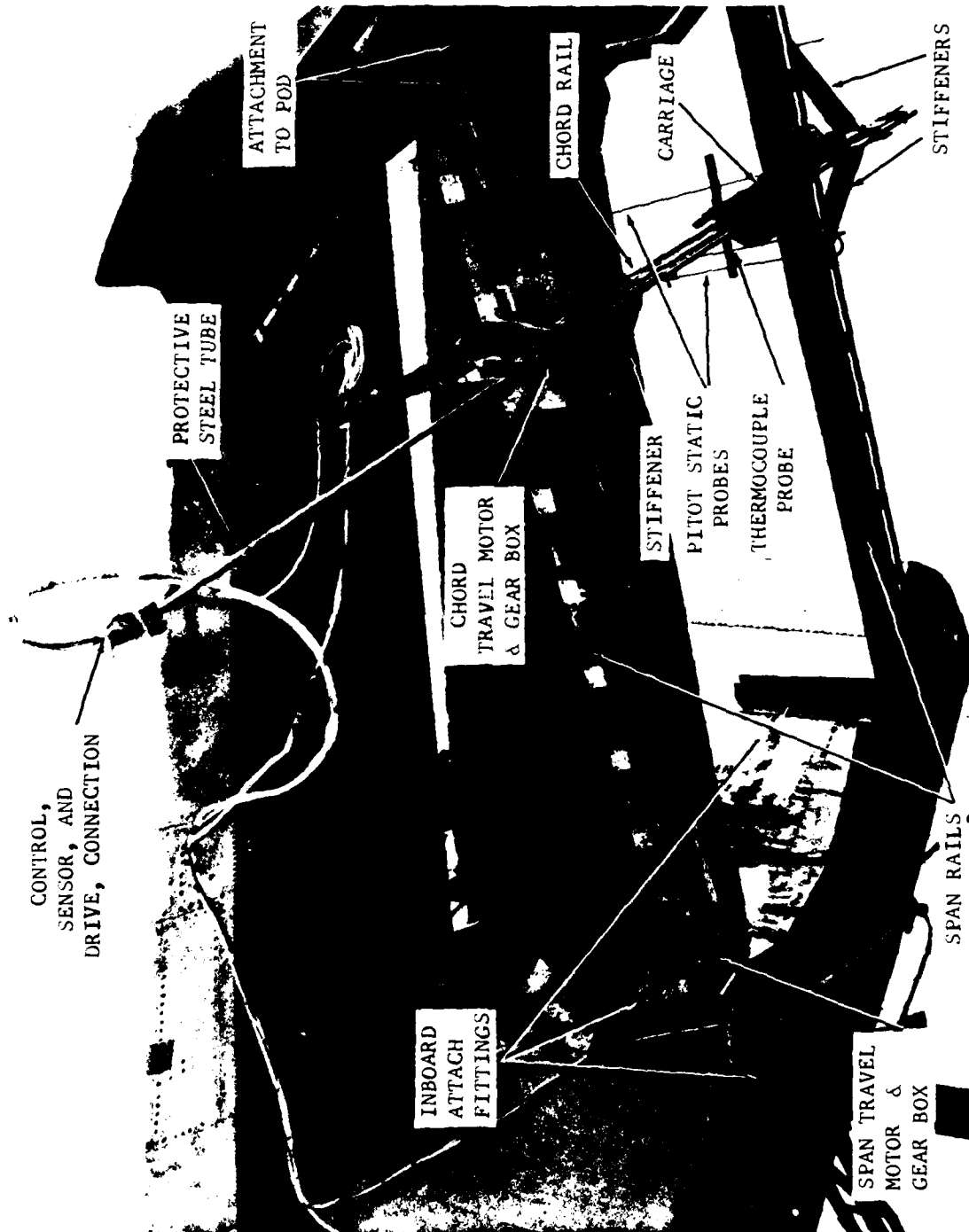
Figure 10-11. Traversing Probe - Installation
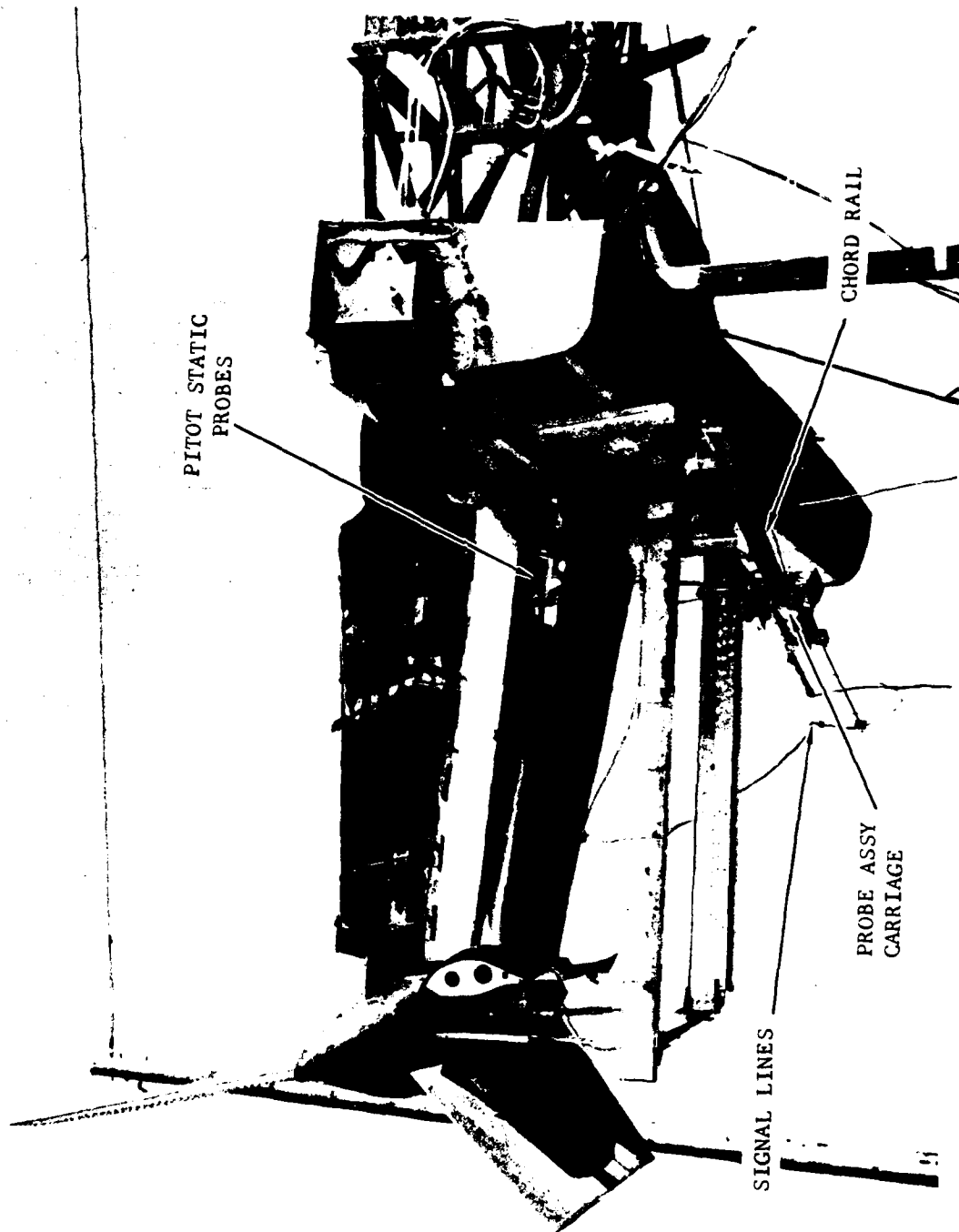
Figure 10-12. Traversing Rake Instrumentation for Center Ejector Jet Growth

Figure 10-13. Schematic - Thermo Lab Process Air System

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COLUMBUS DIVISION LABORATORY

COMPRESSOR OPERATING CHARACTERISTIC

TEMPERATURE = 75°F
BAROMETER = 29.4 IN HG

DESIGN OPERATING
LINE

PRESENT OPERATING
LINE

COMPRESSOR FLOW LBS/SEC

(Y-axis: COMPRESSOR DISCHARGE PRESSURE - PSIG; values 120, 160, 200, 240, 280, 320, 360, 400, 440)

(X-axis values: 13, 14, 15, 16, 17, 18, 19, 20)

Figure 10-14. Columbus Division Laboratory Compressor Operating Characteristics

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close tolerance pressure variations are regulated at the test site by
support personnel through a series of control valves and regulators
located in the test hardware feed lines. Model pressures and flow rates
are monitored by test support personnel at the specific test stand
location as discussed in the test apparatus section of this report. The
high pressure compressor system was utilized extensively during the model
scale centerbody nozzle and augmenter shroud test programs as the primary
air supply system for these models. During the full scale XFV-12A elevon
diagnostic investigation, a series of tests were conducted in the
Thermodynamics Laboratory hazard proof test cell number 1. This test cell
is a reinforced concrete test chamber 17'W x 37'L x 14'H. The test cell
was supplied primary mass flow delivered from the compressor through a
four-inch line. Mass flow rates and test hardware pressures are monitored
by venturi and pressure transducer instrumentation described in the full
scale elevon section of this document.

10.4   TEST FACILITY - WPAFB

10.4.1   General

The test facility utilized in the Coanda segment model and full scale
elevon test programs was located at the WPAFB Aeronautical Propulsion
Laboratory, adjacent to Building 18C. The site, prior to the start of the
test, shown in Figure 10-15, consisted of a concrete slab, screened-in
enclosure and a 12-inch air supply line equipped with a manually operated
gate valve. The air source consisted of two centrifugal compressors
located in the basement of Building 18E. In order to make the existing
facility suitable for testing, an air regulation and flow metering system,
load stand, enclosure and operations trailer were supplied by the
contractor. The modifications to the existing facility were designed to
accommodate testing of either the segment model or full span aircraft
elevon. The test facilities before and after installation of the
enclosure are shown in Figure 10-16.

10.4.2   Test Enclosure

In order to provide protection from the elements and maintain area noise
levels within the WPAFB safety limits, an enclosure, 23 feet by 17 feet
was assembled around the existing sheet metal roof, as shown in
Figure 10-16. The interior of the enclosure was acoustically lined to
attenuate the noise level to acceptable levels. Attenuation of the dump
door noise was accomplished by a rectangular duct and an acoustically
lined cone 18.5 feet high which directed air vertically. Farfield noise
level data were acquired by WPAFB personnel during initial runs to ensure
that the 95 dbA limit was not exceeded at prescribed stations in the
surrounding area. Results of these tests are presented in Figure 10-17.

The four enclosure walls were constructed using 1/2 inch plywood and
2 x 4's; between studs were 3-1/2 inch fiberglass sheets. This was faced
with composite material KC-5-50/150B for sound absorption. Two standard
size doors, 3 feet x 7 feet, were installed in the east side of the
enclosure. A rack of two acoustic panels, 11 feet by 8 feet, were

VIEW FROM PARKING LOT

VIEW FROM PATIO

FIGURE 10-15. TEST SITE UNMODIFIED

SOUND ABATEMENT STACK

EXTENDABLE PANELS

TEST ENCLOSURE WITH SOUND ABATEMENT PANELS

OPERATIONS TRAILER

VENTURI

MODEL

LOAD STAND

CONTROL AND MAIN SHUTOFF VALVES
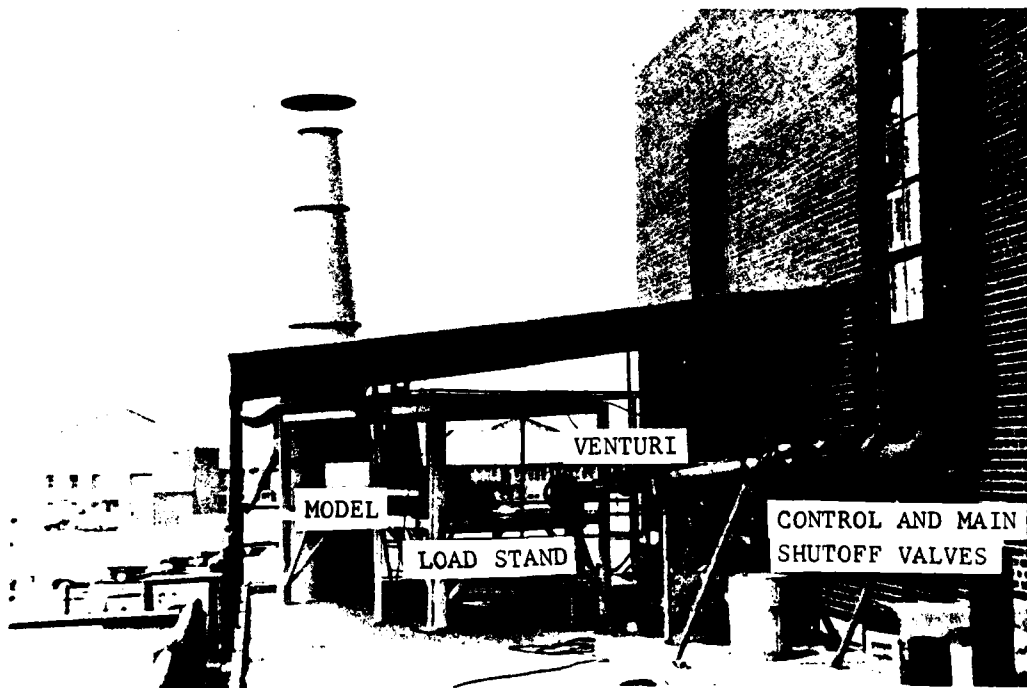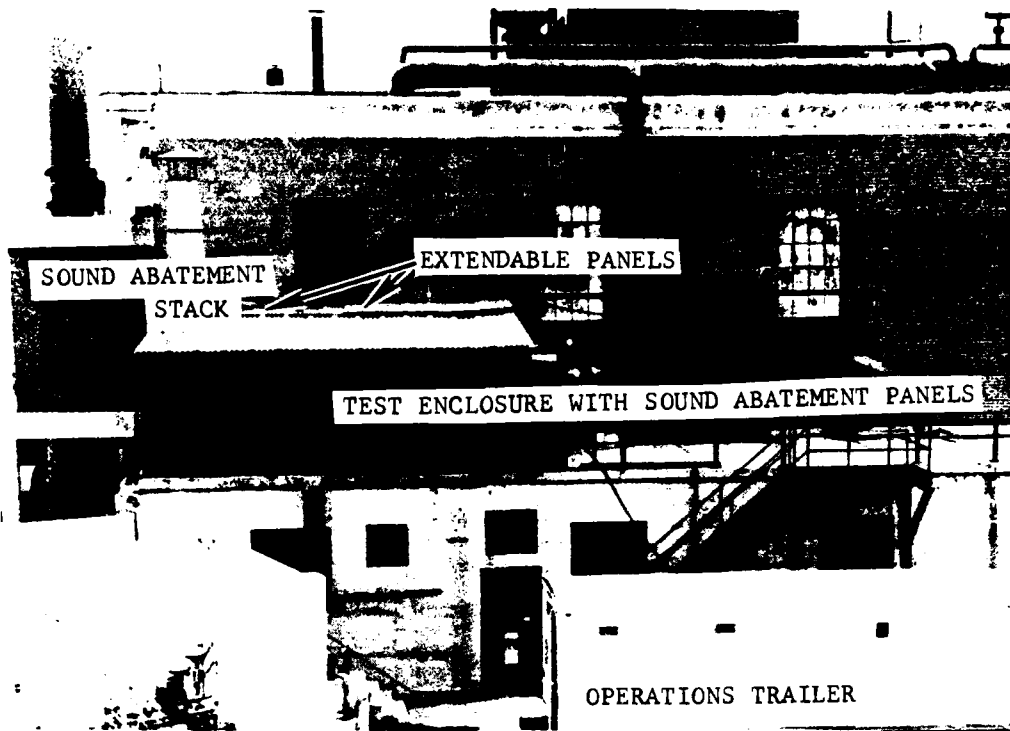
FIGURE 10-16. TEST FACILITIES

TEST SITE -
ELEVON COANDA
SEGMENT MODEL
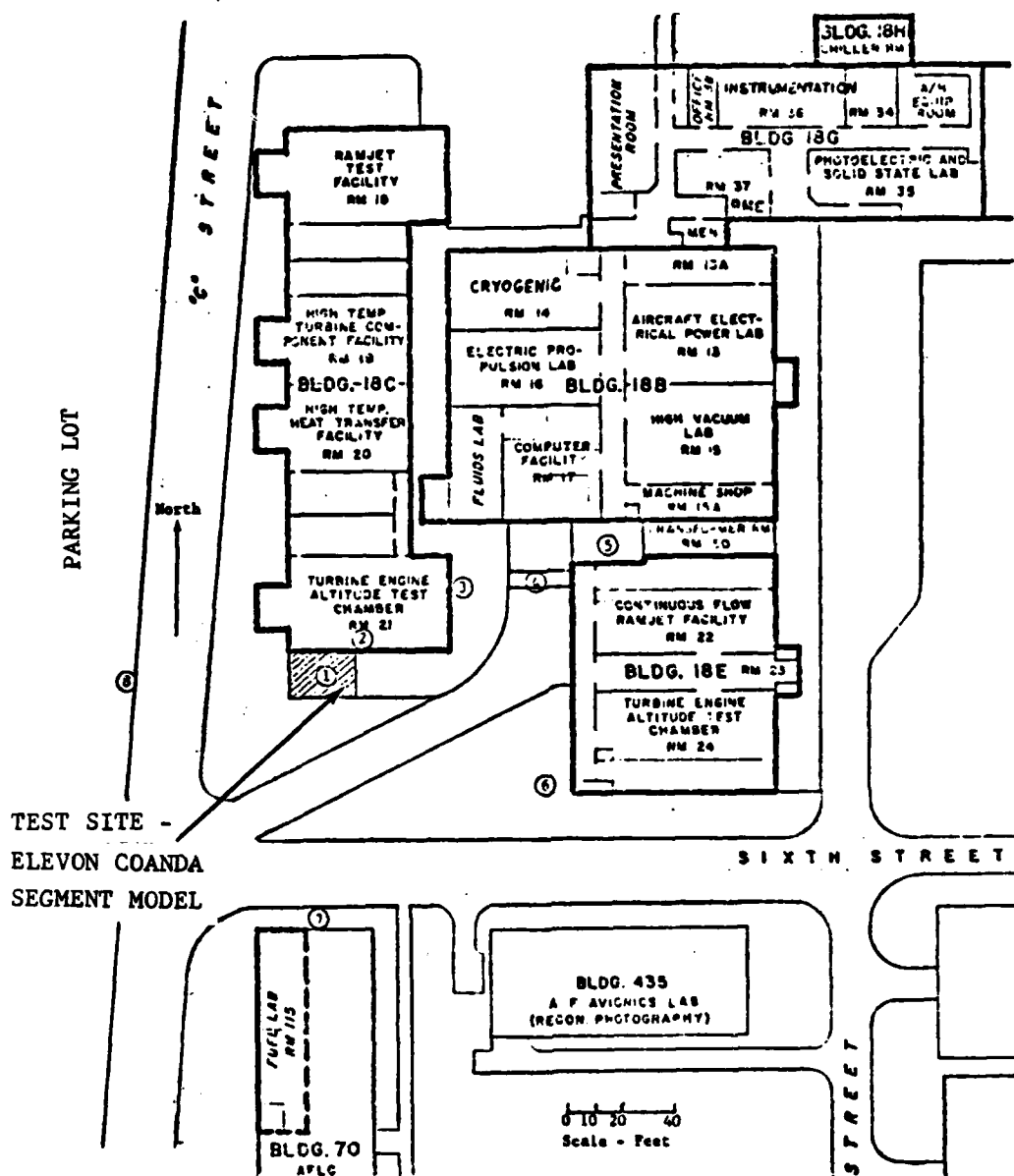
MEASURED NOISE LEVELS

		dbA				dbA
(1)	Inside Test Enclosure	128.0	(5)	At Bldg. 18B - 18E Platform		77.5
(2)	Inside Room 21	84.0	(6)	Stairway Outside Room 24		81.0 (90.0)
(3)	Patio Outside Room 21	79.5	(7)	Lower Level at Bldg. 70		78.5 (79.5)
(4)	At Bldg. 18E Door	79.5	(8)	Sidewalk Across Street		77.0 (78.0)

( ) Test Enclosure Access Doors Open

FIGURE 10-17. TEST FACILITY - NOISE MEASUREMENTS

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assembled and installed above the model to attenuate and prevent reflections of noise within the enclosure (off the metal roof). Nozzle efflux was exhausted through an opening in the northeast corner of the roof during Phase I testing. An additional opening in the northwest corner, nearer the model exit, was provided for Phase II and subsequent testing to reduce enclosure recirculation.

10.4.3   Operations Trailer

The trailer, shown in Figure 10-16, was used as a test control station, work shop and office space. With the exception of the instrumentation sensors and X-Y plotters required for flow surveys, all recorders and signal conditioning equipment were located in the trailer.

10.4.4   Instrumentation and Data

The instrumentation parameters which were recorded and monitored are listed in Table 10-3. These parameters were primarily load stand forces, air supply system and model pressures and jet survey pressures. Selected parameters were displayed as well as recorded for test control purposes. Manometers and pressure gages were located outside the test enclosure and in the vicinity of the model flow control equipment. All end sensors were located either within or just outside the enclosure. Instrumentation control, signal conditioning, power supplies, Digitrend recorder and test equipment were located in the operations trailer.

Printout data from the Digitrend were displayed in voltage levels and were acquired at a rate of two samples per second. This information was tabulated following each test in an order compatible with the data reduction program. The data reduction program was generated specifically for these tests; however, it was based primarily on modifications to portions of existing programs. Voltage levels from each input parameter were converted to engineering units and computations for nozzle pressure ratio, venturi weight flow, C_D, true lift, nozzle isentropic thrust and C_V were performed. Leakage rate and load stand interaction calibrations were incorporated as the model configuration and/or installation was changed.

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Table 10-3 Instrumentation Parameters

PARAMETER	RANGE	SENSITIVITY	DIGITREND CHANNEL	DIRECT READOUT
1. Lift Load Cell (L_1)	0-500 lb	100-lb/v	63 and 78	
2. Lift Load Cell (L_2)	0-500 lb	100 lb/v	64 and 79	
3. Side Load Cell (S_1)	0-1000 lb	100 lb/v	65	
4. Side Load Cell (S_2)	0-1000 lb	100 lb/v	66	
5. Drag Load Cell (D_1)	0-500 lb	100 lb/v	67	
6. Drag Load Cell (D_2)	0-500 lb	100 lb/v	68	
7. Drag Load Cell (D_3)	0-500 lb	100 lb/v	69	
8. Drag Load Cell (D_4)	0-500 lb	100 lb/v	70	
9. Regulator Inlet Supply Pressure	0-100 psig	10 psi/v	71	Pressure Gage
10. Upstream Venturi Pressure	0-100 psig	10 psi/v	72	Pressure Gage
11. Venturi ΔP	0-25 psid	10 psi/v	73	Manometer
12. Model Interface Pressure, P_T	0-50 psid	10 psi/v	74	
13. Model Nozzle Midspan, P_T	0-25 psid	10 psi/v	75	Manometer
14. Model Interface Ref. to Midspan	0-5 psid	1 psi/v	76	
15. Rolling Probe and Gun Probe P_T	0-50 psid	10 psi/v	-	X-Y Plotter
16. Venturi Air Supply Temp.	0-200°F	-	80	
17. Ambient Air Temp. (Outside)	0-100°F	-	81	
18. Traversing Probe P_T	0-5 psid	1 psi/v	-	X-Y Plotter
19. Hose (Tare) P_T	0-50 psid	10 psi/v	62	
20. Air Duct Dump, P_S	0-5 psid	1 psi/v	77	
21. Air Dump Tube Outlet, P_T	0-25 psid	10 psi/v	61	
22. Barometric Pressure				

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11.0 AIRCRAFT PRESERVATION

11.1 GENERAL

At the conclusion of the NASA Langley static and dynamic tests in July 1978, the XFV-12A Prototype No. 1 aircraft was placed in storage in the large NASA flight hangar. Figure 11-1 shows the aircraft as stored. As can be seen, the storage area does not provide a benign environment. Therefore, to provide for this storage, an initial inspection was conducted including cleaning and preservation of the aircraft to inhibit deterioration. This preservation was accomplished in accordance with Contract N00019-73-C-0053, MOD P00040, Item 13, and NAVAIR Instruction 15-01-500. Subsequent inspections have been conducted periodically since February 1979 and approximately every 120 calendar days thereafter.

To implement the initial storage, a checklist was prepared using NAVAIR 15-01-500 and knowledge of detail areas of the aircraft that were subjected to unusal usage during the previous test period. This checklist was submitted to and approved by the contractor's local NAVPRO Quality representative. Table 11-1 itemizes the specific tasks inspected.

Selected items were removed from the aircraft for additional tests at the contractor's plant in Columbus, Ohio. The items included but were not limited to: YF401-PW-400 engine; left-hand canard and supply ducting; left-hand wing augmenters and supply ducting; and selected fuselage parts. All items removed are noted in the aircraft log book.

11.2 INSPECTION/PRESERVATION SUMMARY

The initial inspection/preservation was completed in February 1979 and the second inspection/preservation was completed in June 1979. Very little change from the original storage condition was noted. A general cleaning was conducted and a reapplication of preservative applied to areas which indicated suspected deterioration.

The third inspection/preservation was conducted in October 1979. Accelerated corrosion had occurred during this period. A thorough cleaning of the affected areas and preservative was reapplied.

The fourth inspection/preservation was conducted in January 1980. Because of the previous deterioration noted in the third monitoring period, a more extensive cleaning was conducted. Additional items were removed for inspection and subsequent cleaning. These items were stored as loose items with the aircraft. Preservative was reapplied to the entire aircraft.

The fifth inspection/preservation was conducted in May 1980, and the sixth inspection/preservation in September 1980. Very little change had occured since the fourth period. The fuselage sump tank was opened and inspected during the sixth monitoring.

In February 1981, a visual inspection was conducted and a recommendation was made, and accepted, to the Navy to delay the formal inspection.

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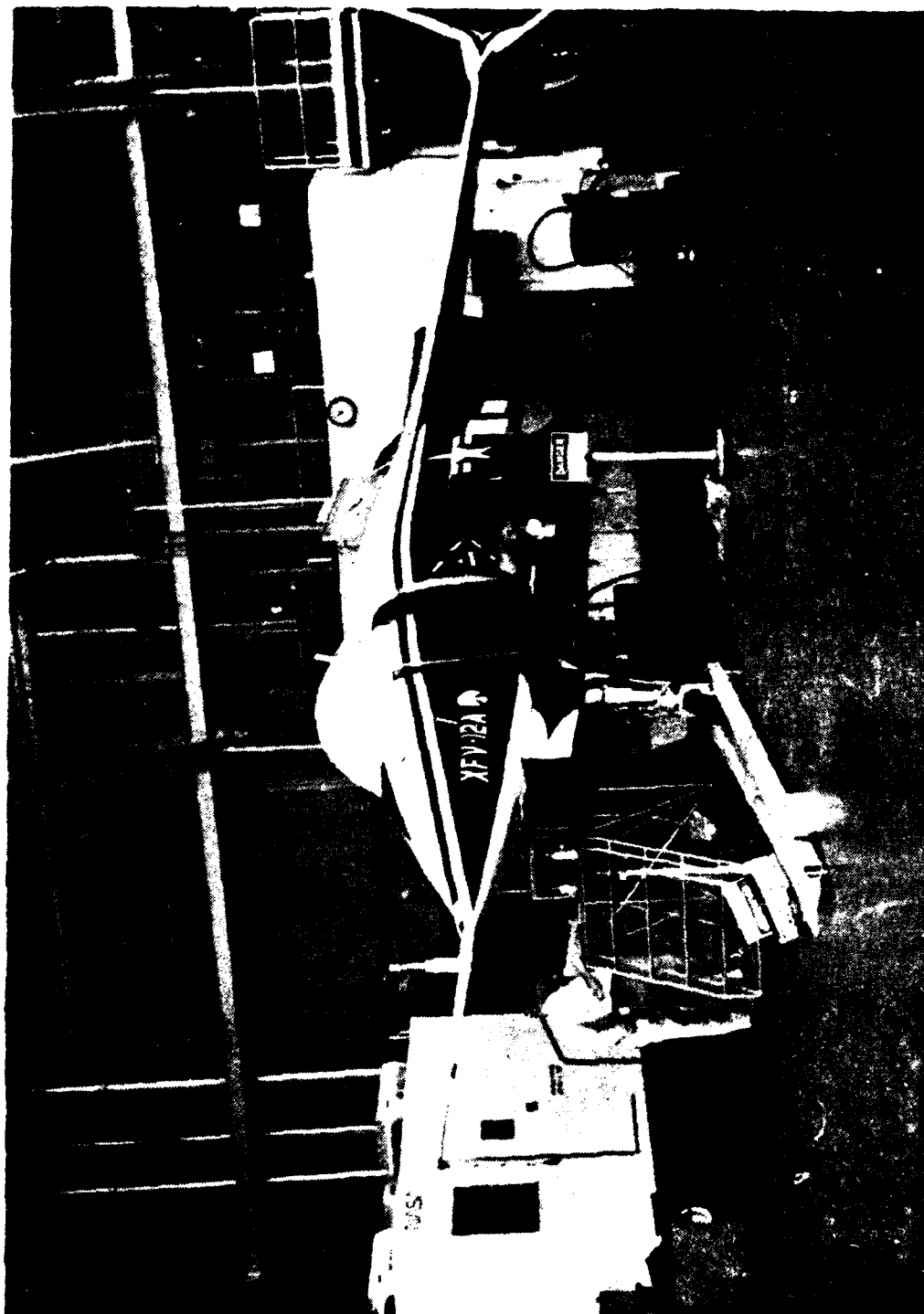
Figure 11-1 XFV-12A At NASA Langley - General View

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Table 11-1 XFV-12A – Long Term Indoor Ventilated Storage – Type A Modified
Ref. NAVAIR 15-01-500

SUBJECT			MECH.	INSP.
General	Clean airplane internally and externally; pay particular attention to areas exposed to engine gases. Treat corroded areas.			
	Lubricate in accordance with applicable instructions.			
Access Doors	Lubricate: clean seals and mating surfaces; dust with talc; coat pins with MIL-C-81309(WP)			
Bellcranks	Inspect; treat corroded areas.	Cockpit		
		Mixer Bay		
		Engine Bay		
		Wing F/C Bays		
		Canard Areas		
		Wing Pods		
Cables	Clean representative section. Inspect for corrosion; clean and coat as necessary with MIL-C-16173 Grade 1	Cockpit		
		Mixer Bay		
		Engine Bay		
		Wing F/C Bays		
		Canard Areas		
		Wing Pods		
Pulleys	Inspect for frozen bearings and corroded cables; treat as required.	Cockpit		
		Mixer Bay		
		Engine Bay		
		Wing F/C Bays		
		Canard Areas		
		Wing Pods		
Canopy and Windshield	Clean and polish with P-P-560 compound and cover with cotton flannel cover			

SUBJECT			MECH.	INSP.
Brakes	Disassemble; clean; inspect for corrosion. Treat as required; reassemble.	L/H Main		
		R/H Main		
Landing Gear Bearings	Remove; clean; inspect for corrosion. Treat as required; repack.	Nose		
		L/H Main		
		R/H Main		
Wheels	Clean; inspect for corrosion. Treat as required. Apply MIL-C-16173 Grade 2 to unpainted surfaces; reinstall.	Nose		
		L/H Main		
		R/H Main		
Tires	Inflate and Clean	Nose		
		L/H Main		
		R/H Main		
Landing Gear Struts	Service; deflate to 1-inch above minimum height. Coat exposed pistons with MIL-D-16173 Grade 2.	Nose		
		L/H Main		
		R/H Main		

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Table 11-1 XFV-12A - Long Term Indoor Ventilated Storage - Type A Modified (Continued)
Ref. NAVAIR 15-01-500

SUBJECT		MECH.	INSP.
Fuel System Lines & Fittings	Clean; inspect for corrosion. Treat as required.		
Hydraulic System	Service reservoirs		
Hydraulic Actuators	Clean; inspect for corrosion. Treat as required. Coat exposed pistons with operating hydraulic fluid. Mixer Bay Wing Controls Canard Controls Landing Gear		
Hydraulic Lines and Fittings	Clean; inspect for corrosion. Treat as Required. N/G Well Cockpit Mixer Bay Engine Bay Wing F/C Bay Canard Area Wing Pods		
Landing Gear	Clean and lubricate; inspect for corrosion. Treat as required. Apply MIL-C-16173 Grade 2 to unpainted surfaces. Nose L/H Main R/H Main		
Axles	Remove wheels; clean; inspect for corrosion. Treat as required Nose L/H Main R/H Main		
Cockpit	Clean and treat corroded areas		
Quadrant	Inspect for frozen bearings and corroded cables; treat as required		
Instruments	Clean lenses - cage gyros - cover instrument panel and consoles with polyethylene plastic sheeting		
Rudder Pedals	Clean; treat corroded areas		
Control Stick	Clean; treat corroded areas		
Seat	Remove; inspect for corrosion; treat as required. Reinstall and cover with polyethylene plastic sheet (NOTE: SEAT & CANOPY NOT ARMED. PARACHUTE STORED AT NAAD)		
Skin Surfaces Exterior	Inspect; treat corroded areas then apply MIL-C-16173 Grade 2 to hinges and unpainted surfaces Fuselage Canard Wings Pods Verticals		
Torque Tubes and Control Rods	Inspect for frozen bearings and corroded areas; treat as required. Cockpit Mixer Bay Engine Bay Wing F/C Bays Canard Areas Wing Pods		
Fuel System Tanks	Defuel and drip drain system; spray cells with MIL-L-6081 Grade 1010 oil.		

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All discrepancies, removals, cleaning, and applications of preservative are noted in the aircraft's inspection record.

The present condition of the aircraft is acceptable for an interim period, but if use within the next two years is contemplated, some inspection should be done on a semi-annual basis. These inspections do not assure the operational capability of the airframe or the systems, but only that corrosion effects have been minimized.

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12.0 CONCLUSIONS AND RECOMMENDATIONS

During the time covered by this report, the XFV-12A program activities were directed primarily toward the development of improved augmenter performance with configurations compatible with high performance aircraft designs. This improved performance was derived from extensive and coordinated analytical and experimental efforts which provided an expanded understanding of augmenter flow parameters, component contributions, and effects of design changes for this type of ejector thrust augmenter configuration.

Program activities and objectives were highly coordinated among the Navy Project Management, the Navy Advisory Committee and the contractor. The resulting pragmatic approach produced excellent results with regard to technology development. Most of the program objectives were achieved. It is considered unfortunate that funding was terminated before these excellent results could be demonstrated with full scale hardware in order to provide a highly visible, practical, verification of this concept for high performance V/STOL aircraft.

Specific conclusions and associated recommendations with regard to future activities follow.

1. Development testing of a full size Coanda/flap surface showed that the quality of flow can be improved by relocating the nozzle away from the feed duct. This relocation, together with an optimization of the Coanda shape, eliminated the flow augularity exiting from the nozzle at the inboard end of the surface in addition to improving the nozzle total pressure distribution along the span. These changes resulted in improved surface flow and velocity coefficients.

    Recommendation: Develop a design criteria, in the Laboratory, using a large scale model to produce quality flow at the inboard end of the centerbody nozzle. Based upon the above, design, fabricate, and validate full scale on the whirl rig an improved centerbody nozzle configuration.

2. Through the use of physical reasoning, mathematical analysis, and carefully controlled test techniques, scale model testing can be utilized in the development of large scale ejector systems. Experimental data have been obtained which provide substantial evidence that model scale ejectors powered by cold air jets provide a close approximation to the performance of full size ejectors powered by hot jet exhaust flows.

    Recommendation: Using scale models and full scale components develop exact scaling parameters for ejectors.

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3. Using a finite difference solution of Reynolds equations, which employs a two-equation turbulence model for closure, a numerical model has been developed, and successfully demonstrated by scale model tests, to accurately predict differential levels of performance related to centerbody nozzle configurations. This numerical model was developed for two-dimensional analysis and does not include performance effects related to Coanda shapes or three-dimensional inputs.

> Recommedation: Incorporate Coanda jet turning phenomenon and correlate with existing experimental data base. Using this revised numerical model incorporate three-dimensional effects and validate by scale model tests.

4. Through the use of systematic testing of model scale and full size components, numerical analysis, and aircraft design studies of configuration restraints, improvement of the thrust augmentation level of the XFV-12A wing augmenter has been increased from the 1.26 level at Langley to 1.64 in the Laboratory.

> Recommendation: Design, fabricate, and test a full scale ejector, of the configuration developed during this effort, on the whirl rig facility using the YF401 engine.

5. Aircraft design studies have integrated the improved augmenter components into a high performance ejector wing configuration that provides reduced supersonic wave drag as compared to the original XFV-12A configuration.

> Recommendation: Using the ejector configuration developed under this effort, modify the XFV-12A aircraft for testing of the ejector wing concept in the NASA-Ames new 80 x 120 ft. wind tunnel.